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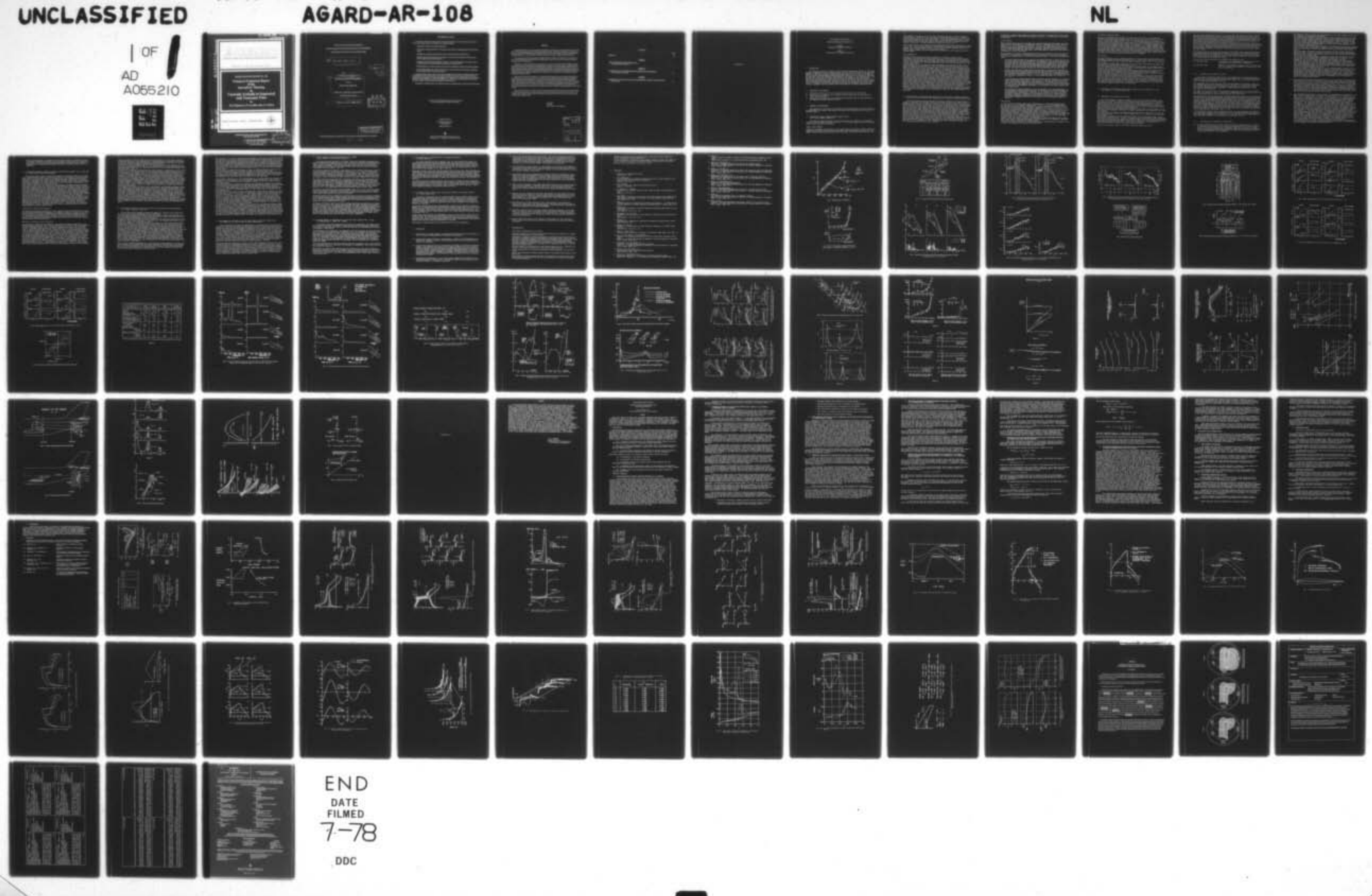

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AGARD ADVISORY REPORT No. 108

Technical Evaluation Report of the Specialists' Meeting on Unsteady Airloads in Separated and Transonic Flow

by

W.J. Mykytow, B. Laschka and J.J. Olsen

NORTH ATLANTIC TREATY ORGANIZATION

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W.J. Mykytow, B. Laschka and J.J. Olsen

This Advisory Report was sponsored by the Structures and Materials Panel of AGARD.

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PREFACE

A Specialists' Meeting on "Unsteady Airloads in Separated and Transonic Flow" was held in Lisbon, on 19 and 20 April 1977, under the sponsorship of the Structures and Materials Panel. The Meeting was divided in two Sessions, the first concerned with "Airframe Response to Separated Flow", the second with "Transonic Unsteady Aerodynamics for Aeroelastic Phenomena". The two Sessions had been prepared in close contact with the Fluid Dynamics Panel.

The evaluation report has been prepared by Professor Laschka, for Session I, and Mr Mykytow, for Session II. It gives a very clear understanding of the reasons for the Meeting to be held, of the content of the papers, and proposes recommendations for future activity.

The Session on "Airframe Response to Separated Flow" gave a survey of the state of the art, through ten papers that presented the most recent theoretical and experimental contributions to this field. Professor Laschka, in his evaluation report, gives a thorough analysis of each of these papers, and points out the main difficulties that still make any buffet prediction hazardous. Most of these difficulties have their origin in an insufficient knowledge of the effect of the Reynolds number on the separated flow pattern on the upper surface of the wing, and in a lack of analytical methods to predict the aerodynamic damping of the first modes of the structure. In his conclusion, Prof. Laschka recommends a strong effort in those two fields.

Eight papers were presented in Session II, on "Transonic Unsteady Aerodynamics for Aeroelastic Phenomena", a field of research which is now very active in the NATO community. Mr Mykytow, as a past member of the SMP, and former Chairman of the Sub-Committee on Aeroelasticity, was well prepared to write the evaluation report of this Session. He delivered in fact the first paper, which gave a survey of the needs of industry for a more accurate prediction of the unsteady aerodynamic forces in the transonic range. His evaluation report summarizes very clearly the output of each of the presentations. In his conclusion, he proposes very detailed recommendations for future experimental, theoretical and computational work, and lays the foundations for a future AGARD cooperative programme, now initiated by SMP.

The Specialists' Meeting on "Unsteady Airloads in Separated and Transonic Flow" has been a success, measured by the quality of the papers that were presented, and by the very lively discussions that took place on this occasion. The evaluation report prepared by Prof. Laschka and Mr Mykytow summarizes this effort and, from the conclusions, helps to prepare future AGARD activity.

G.COUPRY
Chairman,
Sub-Committee on Aeroelasticity

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SESSION I

Technical Evaluation Report on Session I

AIRFRAME RESPONSE TO SEPARATED FLOW

by

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and

W.J.Mykytow

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1. INTRODUCTION

The AGARD Structures and Materials Panel arranged a specialists meating on "Airframe response to separated flow" in Lisbon, Portugal on 19 April 1977 as part of the 44th Panel Meeting on "Unsteady airloads in separated and transonic flow". Chairmen of the meeting were B. Laschka of Messerschmitt-Bölkow-Blohm GmbH, Germany and W.J. Mykytow of AFFDL-USA. The technical evaluation presented here summarizes the results of the contributions of the authors. The titles and authors are given below in the summary of the contributions.
The effects of separated or unsteady flow on military aircraft may lead to failures of primary or secondary structures when exceeding design stress limits or design fatigue loads. During the design and development of military aircraft the aeroelastic and flutter specialists in collaboration with the aerodynamic specialists are therefore confronted with a variety of possible limiting problems, which may constrain the speed-altitude envelope.

2. OBJECTIVES OF THE MEETING

a) Definition and description of the separated and unsteady flow environment

b) Determination of separated flow unsteady pressures and forces by windtunnel and flight test techniques

c) Description and discussison of the analytical approaches used for the prediction of the essential airframe response effects.

3. SUMMARY OF CONTRIBUTIONS

The summaries of the papers of the meeting are presented below and include highlights as presented by the authors. Some general conclusions and recommendations are given in Sections 4 and 5.

3.1 "Unsteady Airloads in Separated and Transonic Flow" by C.L. Bore, Hawker Siddely, UK

This report summarized papers from the FDP Symposium on 'Prediction of Aerodynamic Loading' which was held in the Fall 1976 (ref. 1). It surveys aspects arising from dynamic stall, buffet and separation bubbles.

Dynamic stall aspects

Concepts of transient vortex growth at the leading edge are discussed. Pitching beyond the steady stalling angle is followed by leading edge vortex detachment and affects lift and moment through the downstream convection of the vortex. Pitch rate effects results. Fig. 1.

The frequency of vibration due to the time of the shed vortex to travel the lenght of airfoil chord is roughly 0.27 U/c. The stall flutter mechanism may also be relevant in other problems such as wing rocking and buffet. Flight test measurements (ref. 2) show progressive increase of the wing bending moment at a given angle of attack with increasing pitch rate even below stall. Much of the increase was due to structural inertia effects. Flight test - analysis comparisons ranged from good to fair.

A non-linear lifting line procedure including unsteady wake effects developed by Levinsky (ref. 3) for the near stall instability investigations assumes that each element of wing behaves like a two dimensional aerofoil at the local incidence induced by the time dependent array of bound vorticity and trailing vorticity.
Rolling moment asymmetry and lift hysteresis are predicted by this connecting vortex-lattice method.

Buffet prediction

Different buffet prediction methods are needed for different stages of design.
The Redeker and Proksch buffet onset method (ref. 4) is based on RMS wing root bending examinations, derived from two dimensional fluctuating lift observations and extrapolations to three-dimensional wings. The separated pressure fluctuations are assumed to be constant and synchronised. The method seems useful for light buffet predictions on wings of high aspect ratio.
Another method based on conventional model measurements was proposed by Jones (ref. 5). Wing root bending, torsion moments or wing tip accelerations and the damping of these modes are measured and used to calculate, with the use of linear motion equations, the model excitation forces of the different modes due to separated flow. Aircraft responses in each mode may then be estimated. Improvements of the method could be achieved by the introduction of measured model and aircraft structural and aerodynamic damping. Comparisons of predicted and flight-measured RMS acceleration and damping of the wing bending mode show very encouraging agreement (Fig. 2). The applicability of this method to predict second wing bending and first wing torsion response is however not proved, or may be possible only with additional and more complex instrumentation. The proposed method seems to be useful in the totally separated flow domain, since the difficult and partly not understood problems of motion dependant coupling effects in the excitation forces in the buffet onset region will lead to misinterpretations in the evaluation of the measured strain gauge and accelerometer signals. Careful attention needs to be given to the modal characteristics of the strain gage tranfer functions in the interpretation of modal signals, since cross coupling is possible. It should be noticed that the dampings near the limit cycle torsion instability approach zero and are difficult to measure.
Comments on other papers given at the FDP symposium are given in Dr. Bore's review paper. Concerning separation bubbles, this reviewer states, that the main engineering problem is to predict the conditions which are necessary to cause the flow behind bubbles to be reattached.

3.2 Separated-flow unsteady pressures and forces on elastically responding structures by C.F. Coe and D.W. Riddle, Ames Research Center, NASA and C. Hwang, Northrop Corp.

This paper presented results obtained from several windtunnel and flight test measurements of separated flow unsteady pressures and forces of the F-111A, TACT and F5A. Assessments were made of the effects of elasticity and Reynold's number on the frequency energy content, the phase relationship and the order of magnitude of the unsteady pressures as a function of the investigated wing geometry at high incidences in the high subsonic and transonic speed region. The information is the prevailing result of a systematic buffet study being conducted at Ames Research Center. The objective is to investigate and evaluate buffet response prediction techniques that are based on the measured aerodynamic excitation from windtunnel and flight test and on dynamically scaled model testing. The scope of the buffet research at NASA Ames Research Center is shown in Fig. 3. Some of the research on the F-111A, the TACT (joint USAF-NASA Transonic Aircraft Technology Research Program) and the F-5A has been completed and reported (refs. 7-12). Results of the fluctuating pressure measurement on a 1/6 scale solid steel model of the F-111A variable sweep configuration and of the TACT (joint USFA-NASA Transonic Aircraft, Technology Research Program) 1/6-scale semispan solid steel and aluminium model (Fig. 3) tests of an F-111 configuration with a supercritical wing) performed in the Ames 11 - by 11-Foot Transonic Wind Tunnel were discussed.

Fluctuating - pressure data measured in flight on the thin, low aspect ratio wing of the F-5A and comparison of these data with measured values on a 1/7-scale model are presented in addition.

F-111A results

Eight diagrams have been used to document the fluctuating-pressure development with incidence on the 26° wing sweep F-111A windtunnel model at a Machnumber M = 0.85, a Reynolds number of 12 x 10^6 at one spanwise position y = 0.602. Root mean square pressure at different chordwise locations, power spectral densities and coherence functions of chordwise locations, power spectral densities and coherence functions of chordwise pressure signals, local rms normal force and normal force power spectral densities in the incidence region $0 \leq \alpha \leq 12.3°$ were shown.

The contribution of the F-111A fuselage and inlets to the fluctuating wing flow field was considered to be negligible. From measurements in horizontal tail on and off configuration, is was concluded that the tail had no significant effect on the wing nonsteady pressures.

- Increasing shock strenght and an apparent increasing amplitude of shock wave oscillation with incidence could be observed by the inspection of the root-mean-square pressure fluctuations. The authors point out, that attached, accelerating flows ahead of the shock wave are indicated by decreasing mean static pressures and low RMS levels. Attached flows aft of the shock wave are indicated by positive mean static pressure recovery at the trailing edge and low RMS levels. Separated flows are indicated by negative mean static pressure recovery at the trailing edge and high RMS levels.

- Through the comparison of power spectra and coherence of fluctuating pressures and section normal force fluctuations at different high incidences it was concluded that the separated flow nonsteady pressures show the tendency to be influenced by motion only at certain conditions. Fig. 4, $\alpha = 9.2°$. When separation was widespread on the wing there was insignificant coupling of the aerodynamic forces with motion. Fig. 4. $\alpha = 12.3°$.

- A most significant effect combined with large buffet response occured at an incidence where the shock was located at 35 % chord at 60.2 % span as indicated by a high torsional peak and high correlation in the coherence in the region of the shock wave and near the trailing edge and also high correlation in the coherence at the leading edge of the wing. The authors conclude that the interpretation at these results tends to substantiate the hypothesis that there is a circulation oscillation coupled with the torsional mode.

- With reference to the discrepancies in the frequencies of the narrowband peaks for the integrated excitation spectra compared to the bending and torsion responses, the authors mention a hypothesis of Jones (ref. 5). Accordingly the total fluctuating aerodynamic force could contain a negative spectral peak at the response resonant frequencies because of cancellation of aerodynamic excitation by corresponding in phase motion dependant aerodynamic forces and a positive spectral peak slightly off the resonant frequencies due to the contribution of the out-of-phase aerodynamic forces.

TACT results

In eight diagrams the TACT results of the comparison of a steel and aluminium wing at M = 0.8 and M = 0.9 and Reynolds numbers from 7 x 10^6 to 14 x 10^6 were demonstrated. The RMS values and power spectral densities of fluctuating pressures indicate relativly mild buffet conditions just above the buffet boundary at 9° and M = 0.9 due to a weak shock without corresponding flow separation, whereas at M = 0.8 and M = 0.9 and $\alpha = 12°$ extensively separated flow conditions are present. In general the spectra from the steel and aluminium wings were in good agreement except where motion effects mainly at the second bending frequency, in contrast to the F-111A, influenced the excitation forces. No final conclusion can be drawn as regards differences with respect to motion dependency of the excitation of the aluminium and the steel wing results at M = 0.9, $\alpha = 12°$. The aluminium wing did not show evidence of interaction. Fig. 5.
General trends were found to exist in the coherences in shock wave regions and in separated flow regions. Shock wave coherences are concentrated at low frequencies. Separated flow values extend to frequencies 10 times higher.
Further data analysis will show if spectral and spacial characteristics of pressure fluctuations can be generalized for other configurations.

Influence of Reynolds number

Normal force fluctuations with angle of attack at M = 0.8 in Fig. 6 illustrate the influence of Reynolds number ranging from 7.0 x 10^6, and 10.5 x 10^6 to 14.0 x 10^6. The results of the steel model indicate almost no Reynolds number effect in attached or partly attached flow conditions up to α = 12°. In the completly separated flow condition the normal force fluctuations vary significantly only at the inboard wing sections. Thus static elastic and dynamic elastic effects were negligible. The dynamic pressure varies proportionally with Reynolds number changes in the Ames 11- by 11-Foot Transonic Wind Tunnel, and hence different static elastic effects and dynamic pressure dependent dynamic effect were expected. The discontinuities in the distribution above 12° is believed due to Reynolds number effects on leading-edge vortices and separation boundaries. Some differences were noticed between steel and aluminium model data. They may have been caused by dynamic elastic coupling effects between the pressure and the second bending mode for the steel wing. However, generally both data were in agreement.

F-5A results

Data were presented giving a comparison between full scale flight test measurements and windtunnel fluctuating pressure measurements on a 1/7 scale rigid model for M = 0.75 and for different model test and flight test Reynolds numbers (Re_{mod} = 4.71 x 10^6 and Re_{flt} = 18.9 x 10^6).
For the angle of attack range considered ($8 \leq \alpha \leq 15°$) smaller fluctuating pressures were measured on the aircraft than on the model. (Fig. 7)
The differences are greatest in the lower angle of attack region. At incidences higher than 12°, where total flow separation occured, the descrepancies between flight test measurements and windtunnel are relatively lower.
Unsteady pressure coupling effects with the structural modes were only observed especially in the wing second bending mode and sting balance bending mode at angles of attack lower than 12° in the windtunnel model pressure fluctuations. Flight test spectra showed no peaks corresponding to aircraft vibration modes.
There are different reasons which would explain discrepancies. First the static elastic effect of the full scale wing due to dynamic pressure causes an increase in the local angle of attack of the F-5A in the outer span of about 2°. Within the separation zones, the windtunnel and flight power spectral densities were comparable. Secondly the transient (non-stationarity) aspect of the maneuver, rate of change of angle of attack and the marginal statistical accuracy of flight data could have effects. In the third place the different Reynolds number could be responsible.

3.3 Prediction of transonic aircraft buffet response by A.M. Cunningham, Jr. and D.B. Benepe, Sr., GD/Forth Worth, USA.

3.3.1 Description of the prediction method

The authors present a prediction method for the full scale buffet response. (See also Ref. 8) The method is based on fluctuation pressure data obtained from a rigid scaled windtunnel model. The method requires unsteady aerodynamic forces due to airplane response and natural airplane modes of vibration. A gust response computer program is used to calculate total rigid and flexible aircraft buffet response due to forcing functions obtained from the fluctuating pressure data. The aircraft is aerodynamically balanced and the stability augmentation system disengaged.
Symmetric and antisymmetric responses are combined to form upper and lower bounds for the full scale buffet response. The effects of static aeroelasticity and horizontal tail loads are included.
The investigation is based on unsteady pressure measurements on the wing of a rigid 1/6 scale semi-span model of the F-111 fighter bomber (ref. 7, 13), which are also discussed here in para. 3.2.
The fundamental assumption of the prediction method, that the buffeting forces are not coupled with airplane motions, is due to the choice of the rigid pressure model technique. Since the coupling effects for example of shock oscillation with the torsional motion which can produce strong buffet were known to the authors, they try to evaluate the importance of these coupling effects through their flight test and prediction data correlation.
By necessity Reynold's number effects were assumed negligible.

In Fig. 8 the essential steps of the buffet prediction method are shown.

The generalised aerodynamic forces (Q_{rs}) are calculated with either the subsonic doublet lattice or supersonic Machbox method which allow also the calculation of the interfeting effects of lifting surfaces without modifications in the high incidence region. The generalised buffet forcing function (Q_{rB}) is composed by the forcing function part of the wing, derived through integration of the complex form of pressure spectral densities on the wing and the forcing function part of horizontal tail.
Since tail buffet pressures were not measured, it was assumed that the wake in the vicinity of the horizontal tail due to buffeting pressures on the wing could be predicted with doublet lattice unsteady aerodynamic influence coefficients from the known wing load. Thus the fluctuating pressures and the forcing function part at the tail are derived in the same way as pure analytical interference forces are calculated by the use of the linear potential theory.

The authors establish upper and lower bounds for the buffet response spectra and corresponding RMS values of the predicted wing bending, wing shear, C.G. vertical and lateral acceleration and wing tip acceleration. Since flight test results indicate that the airplane response is usually asymmetric even in "pure" symmetric manoeuvre, both the symmetric and antisymmetric responses are combined.

The upper and lower bound spectra are based on the assumption that

for the upper bound	both symmetric and antisymmetric response spectra are in phase and 100 % active at all frequencies
for the lower bound	symmetric and antisymmetric spectra are 100 % active and 180° out of phase or the symmetric or antisymmetric spectrum is active only.

3.3.2 Comparison with flight test results

The validity of prediction method and of its capabilities is investigated through comparison with flight rest results. The predicted results are in addition correlated with results derived from the prediction method of P.W. Hanson (ref. 9), Mullans and Lemley (ref. 14) and Butler and Spavins (ref. 15).

The comparisons with flight test data, which were presented by the authors in refs. 10, 13 are demonstrated for wing sweep positions 26°, 50° and 72.5° for different Machnumber and altitude. The authors first illustrated an improvement of agreement with flight test results applying the static aeroelastic correction based on the ratio of rigid and flexible lift coefficient.

The upper and lower bounds were verified through comparison with Ref. 10 and 13 flight test data. For example, in Fig. 9 a comparison is demonstrated for the right and left wing tips accelerometer result, which confirms the upper and lower bound.

From the comparison of the influence of different assumptions concerning the introduction of buffeting tail loads as demonstrated for example here on the results of the wing shear in Fig. 10, it was concluded that the horizontal tail was important in the prediction of airplane buffet under conditions well beyond buffet onset. It also appeared that the concept for estimating the tail buffet loads was correct, however, the predicted tail loads were comparatively high. It is believed that this effect would be compensated by taking into account the displacement of the wing wake. The final version of the prediction method takes into account the mentioned aeroelastic correction and the most refined procedure, with total airplane aerodynamics and 1/2 estimated tail loads. The decrease in buffet intensity with sweep as well as the increase of the intensity with incidence and Machnumber is well predicted with the method. (Fig. 11, 12).

In the supersonic case for the wing sweep 50° and M = 1.2 the prediction was restricted to a wing alone simulation due to limitations on the supersonic unsteady aerodynamic calculations. Again, compared to the results at M = 0.85, the decrease in buffet intensity with increased Machnumber is well predicted by the method. (Fig. 13).

3.3.3 Discussion of the capability of the method

a) The over prediction of most of the flight test data by the upper bound, is shown on a frequency of occurrence plot for the wing tip accelerometer and the wing shear. (14 of 38 flight test points fell for example between 20 % and 30 % of the upper bound for the wing tip acceleration prediction). The authors therefore conclude that it is highly improbable that both types of the symmetric and antisymmetric motion could be excited 100 % at all frequencies.

The symmetric motion would be more representative of the total response in the case of interest which was symmetric maneuvers.
The correlation of symmetric predicted response from the current method with flight test data are then shown together with similar results obtained by Butler and Spavins for a second aircraft, by Mullans and Lemley for a third, and by Hanson for the same aircraft investigated by the present authors. Butler and Spavins' method makes use of measured aerodynamic damping and response both as a function of angle of attack from a rigid windtunnel model to predict flight buffet response on the first symmetric bending mode. Fig. 14.
Mullans and Lemley's method uses all symmetric and antimetric modes equally weighted and a constant value of aerodynamic damping for each individual mode determined analytically. The forcing function is determined from rigid pressure model measurements.
Hanson utilised a dynamically scaled elastic model to obtain buffet response data which was scaled up to full scale with a technique similar to Butler and Spavins. However he used a dominant mode damping ratio to scale the wide band RMS windtunnel data rather than a single mode.
The results of Mullans and Lemley's prediction are extremely conservative, mainly due to equal use of symmetric and antisymmetric modes, use of a simple viscous damping, and ignoring aerodynamic stiffening and aeroelastic effects.
Compared to the results of Butler and Spavins and of Hanson the current method yields less scatter and is more conservative. The difference in conservatism is felt to be partly due to the use of linear aerodynamic damping forces for the current method. Damping results for windtunnel data shown by Butler and Spavins indicate that aerodynamic damping for the fundamental wing bending mode increased by about 30 % over the attached flow value.
On the other hand as shown by trends in Fig. 14, Reynolds number effects are probably more important at higher angles of attack as indicated by an increase in conservatism of the current method at high response levels. The result would be expected based on the Reynold's number effects shown by John (ref. 16) and Butler and Spavins, where the buffet forces are shown to decrease with increasing Reynolds number effects. The authors conclude that Reynolds number effects should cause predictions based on windtunnel data from a scaledmodel, to be conservative by some as yet indeterminable degree.

b) The flight test data scatter may be primarily due to the effects of manoeuvre transients, which will lead to variations in free stream dynamic pressure and Machnumber of the order of 5 % to 10 %. One factor of particular importance is the effect of pitch rate on buffeting forces. It is shown in Fig. 11 at = 15.6°, that where the pitch rate was high the flight buffet data were lower relative to prediction. Thus during a manoeuvre pitch rate could be responsible for significant scatter in the flight test data. A statistical approach is therefore recommended, because of the scatter due to the uncontrollable variables that affect the flight data. A frequency distribution of occurrences should be established in order to better define the buffet characteristics for any given airplane.

c) The unconservative prediction of wing torsion, as demonstrated as exceedance of the upper bounds by wing torsion in the low wing sweep case at M = 0.7 and M = 0.8 (Fig. 11, 12) is believed to be a result of the coupling of wing torsion and shock oscillation. The mechanism of coupling was not accounted for in the discussed prediction due to the state of the art in the unsteady aerodynamics. The coupling occurs primarily at low wing sweeps and has been described previously by the GD authors (ref. 10). The shock-torsion coupling, as described by the authors, is attributed to the unstable nature of the primary shock on the upper surface of the wing when it is located near the local crest of the airfoil. A small increase in incidence will cause then the shock to jump from just aft of the crest to just forward. This process results in a loss in lift due to a larger high pressure area behind the shock without any appreciable increase in lift due to incidence, leading to a decrease in lift curve slope. This well known transonic phenomenon occurs long before $C_{L_{max}}$. The basic mechanism is established for large shock excursion with small torsional motions when the wing is at values near the transonic lift curve slope anomaly. A forcing function is produced which is 180° out of phase with the torsion mode, but phase lag will be introduced due to unsteady and pitch rate effects. The aerodynamic damping due to torsional motion should decrease to near zero and torsional response increase accordingly. The response tends to reach a limit cycle because the maximum force is limited in this instability.
It is concluded that, in order to properly account for this effect, the unsteady aerodynamics used to calculate response induced loads would have to include the presence of imbedded moving shocks in the flow field.

Finally the importance of considering buffet fatigue damage on secondary structure is discussed. Secondary structures are not designed to carry high load like primary structures, but they are located in relatively high load buffet areas. A simple survey method for checking the secondary structure at critical stress points is recommended.

3.4 "The Dynamic Response of Wings in Torsion at High Subsonic Speeds" by G.F. Moss and D. Pierce, Royal Aircraft Establishment

The authors mainly discuss a specific phenomenon of unsteady boundary layer separation closely related to buffet, the torsional buzz and its mechanism, occuring on clean swept wings at high subsonic speed and at supercritical flow conditions as the buffet boundary is penetrated. This higher order structural primary torsion mode is studied on the basis of windtunnel test results by the RAE on virtually "rigid" and flexible models having what the authors call more realistic static and also dynamic deformations as regards their extrapolation to flight conditions.
They intend to prove that the intensity of the torsional phenomenon involves interaction of structual response with flow conditions as demonstrated by flexible models and that windtunnel models of solid metal construction even when the frequency and mode shapes are fairly well represented are not appropriate to show reliable torsional effects. In addition they suggest that torsional buzz can be advantageous in some respects as regards the overall manoeuvre performance of an aircraft as fas as the pilot's dynamic environment and effeciency are concerned and which are mainly affected by the lower frequency response of the structure. But, the authors do not exclude limit aircraft fatigue load considerations if the amplitudes become large in the high frequency torsional vibration.
The authors discuss the use of different tools of the aerodynamic and aeroelastic specialist and present data like quasisteady interpretations of steady pressure distributions, local and total lift and moment developments with incidence, unsteady aerodynamic considerations on stochastic excitations due to boundary layer separation at trailing and leading edges at the wing tip region, and damping evaluations. The RAE specialists postulate the mechanism of torsional-vibration or buzz at supercritical flow conditions and demonstrate it to be a sustained high but limited amplitude oscillation similar to single degree to freedom flutter or buzz.

Indication of torsional buzz problems

The described quasisteady reflection on static spanwise pressure distributions and local lift and moment coefficients represents a more realistic method to indicate in the buffet boundary penetration domain the appearance of possible instability problems concerning the bending and torsion mode than the classical "kinkology" based on abnormalities in total lift and moment development with incidence. The latter approach smears out local-effects, which especially are important towards the wing tip, and will therfore perhaps lead sometimes to misinterpretations.

Prediction of severity of buzz vibrations

The RAE authors investigate the severity of the structural response conditions in the buffet boundary penetration region through comparison of results on three wing models with the same aerodynamic design. Fig. 15 shows the characteristic data of the rigid model (577/A1), the model which simulates the static deflection (577/Flex2) and the carbon fiber or scaled frequency model (2070). (See also Figure 18 for non-dimensional characteristics) Relative to the nominally 'rigid' model the 'static deflection' model was about 1/10 as stiff in bending and 1/8 as stiff torsion, and the frequency scaled model (2070) 1/2 as stiff in bending and 1/6 as stiff in torsion. The nodal lines of the primary torsion modes were almost identical in the wing tip region. The 'static deflection' model was tested at two leading edge sweep position 27,2° and 42°. The comparison of the spectra of two unsteady pressure signals (Fig. 17) on the 'static deflection' wing model, one near the leading and the other at near trailing edge, at two leading edge sweep positions and in the incidence region of about $4 < \alpha < 10°$ for the Machnumber M = 0.75, 0.85 and Reynoldsnumber 1.26×10^6, indicates very strong harmonic response at the primary torsion mode only in the 27.2° sweep position at about $6.5 < \alpha < 8.7°$. In the 42° swept back case the pressure signals did not show any large harmonic response in the torsional

mode. The effects on the 42° swept back wing are regarded as a fairly normal pattern of excitation behaviour in the classical case of buffet response. In contrast, the effects on the 27.2° swept back wing are regarded as incipient torsional buzz effects, produced by the shock moving in sympathy with the torsional response.
The severty of the structural response of the torsional buzz is illustrated by the comparsion of acceleration levels (Fig. 18) and corresponding torsional amplitudes and apparent aerodynamic damping on the 'deflection scaled' (511/Flex 2) model and frequency scaled model (2070) (Fig. 19).
As shown for the results of both models the strong build-up of amplitude in torsion starts at the point in the incidence range at which the aerodynamic apparent damping starts to become negative. The comparatively very high torsional amplitude at an incidence of about 8° on the 'deflection scaled' model, is of particular interest concerning the effect on fatigue life. Although scaling to appropriate conditions in flight cannot be done with confidence, the peak value of $\pm$ 0.56° on the deflection scaled model is thought likely to be at least of the same order as that on the aircraft, since a smaller ratio of structural damping to aerodynamic damping is likely in flight. It is concluded that high-speed windtunnel models of conventional "rigid" construction are unlikely to show this torsional buzz phenomenon or be useful in making predictions of the onset boundaries and amplitudes involved. It is proposed to overcome the restrictions by the use of comparatively simple robust 'pseudo' aeroelastic models with appropriate or compromised structural dynamic characteristics.
With respect to the pilot fatigue or pilot efficiency in the considered incidence region it is interesting to note that there is a strong relation between the development of torsional and bending mode amplitude and damping (Fig. 19). The rapid growth of the amplitude and the combined reduction in damping in the primary torsion mode up to buzz coincides with a reduction in the amplitude and an increase in damping of the primary bending mode. The development is just opposite as the incidence is increased. Concerning the pilot who is sensitive in the low frequency region the suggestion is made that the occurrence of buzz in a high frequency mode can be beneficial due to the reduction of low frequency amplitude. The high frequency might be of concern relative to structure and equipment. The extrapolation to flight conditions of the concludions drawn cannot be done with confidence since a comparison with flight test results was not available. The authors propose therefore a well planned flight-tunnel comparison to explore the phenomenon properly.

3.5 Evaluation of vibration levels at the pilot seat caused by wing flow separation by J. Becker, MBB and K. Dau, VFW

In this article two examples of prediction methods of vibration levels expecially at the pilot seat caused by separated flow are presented.
The first deals with the extrapolation of wing root bending moments from windtunnel testing. Low speed measurements on a wing model with strake in a clean and high lift configuration including leading edge blowing are analysed.
The second example demonstrates the result obtained by the prediction method based on measurements of fluctuating pressures on rigid models for two configurations with 25 and 45 degree wing sweep in the high subsonic region.
In the first example it is demonstrated that the effect of a strong leading edge vortex generated by the strake on the wing, which creates additional lift up to high angles of attack, leads to very low vibration levels as far as the wing root bending and torsion (Fig. 20) of the clean wing are concerned. The high lift configuration shows, and that is the most interesting point, considerably higher vibration amplitudes in the wing root bending and wing tip acceleration signal in the bending frequency response at the condition with blowing, that is at higher energy flow conditions compared to the condition without blowing and the clean wing configuration at maximum lift conditions. Beyond maximum lift the vibration levels are very small. The extrapolation of the measured wing root bending signals to the vibration levels at the pilot seat was not possible. Through the Mabey method would propose this, the extrapolation to the real aircraft would require buffet criteria which are independent of the windtunnel unsteadiness. Further more the total aircraft elastic behaviour, for example the fuselage stiffness and structural damping in neglectad completely in such an extrapolation. Therefore this investigation could only be used for trend studies.

The pilot seat vibration investigation of a variable wing sweep aircraft in the high subsonic-transonic region demonstrates the results derived by a method based on the measurement of fluctuating pressures on a rigid model. The influence of wing sweep, Machnumber, the influence of external store and fueled and empty wing was studied to indicate the effects on the buffet intensity at the pilot seat.

The assumption of the rigid pressure model technique is, that the full scale flow pattern and flow energy is similar to the model one. The assumption of flow similarity in not necessarily true, because the buffet excitation forces, due to boundary layer noise, random and periodic vortex shedding and shock oscillations, are a function of Reynolds number and vibration amplitude. The windtunnel and flight Reynolds number differences will give rise to an uncertainty. The deflections of the full scale wing will be disproportionally larger than those of the stiff wing model and therefore might not only change the separation pattern, but also introduce motion dependent changes in structural and aerodynamic damping forces.
The uncertainty of predicting the angle of attack at which maximum lift occurs also exsists in the determination of the angle of attack for buffeting predictions.
The effect on the flow separation buffet pattern on the wing caused by large static deflections at high wing loading could be compensated to some extent in the rigid pressure model technique by assigning those buffet pressures to the static angle of attack that corresponds to the rigid model angle of attack of the same magnitude.

The aerodynamic damping due to the large deflections of the full-scale wing could be represented as a first approximation by aerodynamic forces derived from linear theory. A correction of the theoretical linear aerodynamic damping by the use of the lift curve slope is proposed.
The measured results at 25 degree wing sweep of the fluctuating pressures and accelerations revealed strong increase in the RMS torsional acceleration, without strong increase in the RMS bending acceleration and in the RMS pressures with increasing incidence. (Fig. 21, 22) Since there was no indication of motion coupling in the fluctuating pressure spectra, decrease of aerodynamic damping in the torsion mode is therefore the possible explanation.
The exitation forces in the 45 degree wing sweep are of same order of magnitude as in the 25 degree case if the flow is not totally separated. In case of locally total flow separation as observed at M = 0.8 around   = 0.67, where suddenly a lift curve anomaly is observed, it is believed that the aerodynamic damping of the torsion is almost zero. During this process the fluctuating pressures are very small in the separated region. In this connection stall flutter aspects are discussed. (Fig. 23) The observed "instability" is thought to be caused by resonant response of the torsion mode with the Strouhal eddy shedding frequency which is believed to be proportional to the projection of the local wing chord in streamwise direction and extremely dependent on Reynolds number in analogy to the effects on circular cylinders.
The comparison of the dynamic response with the results of the prediction methods yields valuable conclusions. The power spectral densities of the measured accelerations showed response of the wing in the first, second bending and first torison mode. The magnitude of the first wing bending is predicted well by introduction of total aerodynamic damping by linear theory for this mode, whereas a reduction in the introduction of linear aerodynamic damping was necessary to predict the measured accelerations of the second bending and first torsion mode. (Fig. 24) It is concluded that this dynamic response method based on rigid model unsteady pressure measurements with some necessary modifications of the aerodynamic damping of different modes, is suitable to predict pilot seat accelerations.

3.6 Measurements of Buffeting on Two 65° Delta Wings of Different Materials by D.G. Mabey and G.F. Butler, Royal Aircraft Establishment

Buffet measurements on steel and magnesium delta wing models having the same frequencies but different responses and total damping ratios were made to show that the non-dimensional buffet excitation parameters appropriate to the first bending mode could be derived from measurements of different responses and total damping ratio. The scaling law was substantiated. The non-dimensional parameter was nearly the same for both wings and was independent of Reynolds number except at low values. (Fig. 25) It increases substantially after vortex breakdown. Above 18° and M = 1.4 the magnesium wing oscillated violently, possibly due to shock wave oscillations and aeroelastic distortion effects. Aeroelastic distortions did not influence mean force measurements on the two wings at M = .35 and .70 and probably the buffet parameter determined from these tests. However, they could influence other results and always require evaluation on flexible components and models. Cryogenic wind tunnels indicate possibilities for separately evaluating Reynolds number and aeroelastic distortion (kinetic pressure) effects.

Aerodynamic damping changes were noticed for the lighter magnesium wing (Fig. 26). Aerodynamic damping ratio measurements agreed with slender wing theory predictions. While it appears possible to determine the buffet excitation parameter with accurate measurements of response and total damping, extractions of structural and aerodynamic dampings may be difficult thereby limiting accuracy in computing the buffet response of flexible aircraft. Additional experimental and analytical studies are needed to determine the relative magnitudes of structural and aerodynamic damping in the total damping. This method has appeal for simple modes. Data can be provided in the earlier aircraft design stages. The method could be used for the torsional mode.

3.7 Dynamic Loading of Airframe Components by C.G. Lodge and M. Ramsey, British Aircraft Corporation

The authors discuss prediction of unsteady loading for secondary structures which could be influenced and designed by such loads. For similar aircraft the aerodynamic and inertia loads and attachment responses may be obtained by scaling. Slat extentions can reduce dynamic attachment loads. Spoiler attachment loads vary linearly with spoiler anlge below the wing anlge at which maximum loads occur. Above this angle wing separation spreads over the spoiler. (Fig. 27)

Wing torsion response was measured in the moderate buffet response during high incidence load measurements of an earlier BAC (MAD) aircraft at M = 0.9. (Fig. 28) The authors discuss the possibility of an association with alternative flow states. (Figure 29) A similar torsional response was noticed on a new prototype. (Fig. 30) Lower sweep angles were more critical. Further measurements are being made to evaluate stability and fatigue aspects. (See also comments by other authors on torsional response and instability).

The authors question the applicability of unsteady root bending tail measurements from rigid models of highly swept-back wing concepts because of significant flexible mode response and non-simulation of modes and frequencies. Figure 31 shows data from model tests. Taileron loads in the fundamental mode respond to wing buffet excitations at low Mach number and sweep angles. At higher angles the taileron response is due to both tail buffet and mild wing buffet. Increased wing sweep increases tail buffet significantly above M = 0.7 - 0.8. Maneuvering wing slat significantly reduce taileron loads.

The authors also discuss external store attachment loads for M = .85 - .95 due to shock induced flow separation and the alleviation with the use of vortex generators. External stores significantly increase tail root loading variation with increasing lift coefficient. Model tests confirm this trend with incidence.

Downstream excitation from excrescences (spoilers) and cavities are discussed. Estimation of the separation scale and unsteady pressure amplitudes are difficult judgement matters in the earlier design stages. Typical excitation shapes for shallow configurations are given. (Fig. 32) Airbrakes and deeper cavities will be affected by periodic flow effects which will require consideration in design.

In summary, the authors point out various unsteady aerodynamic and aeroacoustic loadings. The early prediction of the loadings cannot be made with high confidence for separated flow, but the need for such predictions especially tail loads, increases. Model tests can provide much useful data. The aeroelastic (flutter) - buffet model approach appears very suitable for buffet load prediction including the wing torsional response phenomenon. However, load restrictions on the aeroelastic model could limit investigations. Provision of detailed pressure distributions from rigid models might be more economic and provide more insight. Dynamic tail load estimations from direct measurements on (semi) rigid models of highly-swept-wing and tail models are open to questions.

3.8 Airframe Response to Separated Flow on the Short Haul Aircraft VFW - 614 by H. Zimmermann and G. Krenz, VFW - Fokker

The authors discuss the determination of (turbo) jet boundaries, jet induced structural loads and the large disturbance, intermittant jet flow on structures outside the jet boundary. Horizontal tail buffet may also be caused by flow disturbances far outside a wing spoiler "wake region".

Jet boundaries are not stationary. The magnitude of the mean velocity is not a good indicator of intermittant turbulent flow occurences. (Fig. 33) Individual disturbances in the edge region move at speeds far above values measured by pitot tubes. (Fig. 34) Buffet occurred on the horizontal tail at full power even though pitot rake measurements inferred low aerodynamic loads since these indicated that the wake was below the tailplane (Fig. 35). Tail buffet vibrations decrease with decreasing engine speed and increasing forward velocity. They are influenced by jet deflections and increase with increasing crosswinds. Dynamic loads were minor but passenger comfort was improved by lowering the engine idle speed.

Pitot rake measurements for the wing spoilers would also erroneously infer low horizontal tail vibrations (Fig. 36). These were not acceptable to passengers. Inner flight spoilers are not used in flight now.

The authors also discuss large differential tail moments due to wing wake excitations during stall, changes in wing nose to improve stall characteristics without significantly affecting buffet boundaries, and heavy buffet produced by small leading edge flap cutouts. Partial covering by a thin plate with aerodynamically poor leading edge resulted in a buffet free flap.

3.9 Tail Response to a Propeller Flow on a Transport Airplane -
L. Chesta, Aeritalie

The author addresses the tail responses (Fig. 37) from discrete high power, unsteady wake excitations of propeller blades; their functions of tail flow conditions; their occurrence and duration as a function of airplane operating conditions, and the tail response and fatigue life. The tail vibrations are a function of power (high pressure turbine speed, N_g and aircraft speed (Fig. 38) and are a maximum when the flow around the tail is parallel to it. (Fig. 39 and 40)

Questions of fatigue life arose for the balance tab and trim tab for realistic operating conditions and this concern was substantiated by ground fatigue tests. Partical and efficient changes to the trim tab (3rd hinge) and balance tab (3rd hinge and leading edge skin reinforcement) and an increase in the elevator torsional mode frequency by changing the elevator mass balance distribution. The tail frequencies now lie in between those excited by main operating conditions. Stresses were reduced and fatigue life increased acceptably.

Tail excitations from unsteady propeller flow are still a problem and their magnitudes cannot be sufficiently well predicted in earlier design calculations with confidence. The process therefore still relies on later flight, vibration and fatigue testing procedures and associated analyses. In the present case, efficient fixes were found but in other cases maintenance expenses and retrofits could be costly.

3.10 A Program System for Aeroelastic Calculations Applied to the Viggen Aircraft
by V. Stark, Saab Scania

The paper is interesting since it outlines Swedish flutter analysis methods, makes comparisons with results from the zero order frequency airload (Pine's) method where aerodynamic damping is neglected, and reports on results approximating the effects of leading edge vortices at angle of attack using the "Pine's" method. Those methods are applied to the Viggen canard-wing combination where three dimensional and interference effects are important.

Generalized airforces and flutter characteristic computed by the SAAB Polar Coordinate Method are in good agreement with those computed by the US AFFDL using the doublet lattic method. Swedish results further show that flutter speeds computed using the p, p-k and V-g methods agree. The critical modes are first wing bending and fuselage bending. The "Pine's" approach yields a flutter speed within 4 % of that computed by the more elaborate methods. (Fig. 41)

Leading edge vortex effects are approximated by using factors based on local lift curve slopes obtained from steady flow measurements. Factors are two (2) for outboard wing stations. (Fig. 42) This reduction, while somewhat higher than indicated by some experimental results, seem to be a plausible estimate. The effect deserves more study and more quantitative test and analytical information is required.

The margins of safety against flutter were high in all cases investigated.

4. CONCLUSIONS

4.1 The meeting on airframe response to separated flow brought forth most important results of recent investigations by the different contributors.

4.2 The methods, though different in the approach, to predict airframe response to separated flow indicate in general similar problem areas as far as several physical effects are concerned.

4.3 Problem areas arise partly due to insufficient knowledge of the effect of Reynolds-number on the separated flow pattern on the upper surface of the wing at high incidences especially in the buffet boundary penetration domain, and to some extent due to the lack of analytical methods to predict aerodynamic damping of different structural elastic vibration modes in the partly separated flow on airframe primary and secondary structures. Consequently confidence in the prediction of motion dependent coupling effects in the excitation forces including the static aeroelastic influences is limited.

4.4 The adequate representation of real flight partly separated flow conditions is in addition difficult to achieve due to the Reynolds number and corresponding dynamic pressure simulation in available windtunnels.

4.5 The various approaches involving measurement - and combined analytical prediction - measurement of wing bending and wing tip acceleration, or the measurements of unsteady pressure on conventional models, or use of static aeroelastic, or partically dynamically similar models have their own advantages as regards the cost and the simulation of actual physical aspects. These methods should be applied by choice depending on the actual project phase, the problem and its severity.

4.6 The method based on the estimation of the excitation force based on measurements on conventional rigid models and the use of measured flight test structural and aerodynamic damping shows until now reasonable results as far as the prediction of first wing bending response is concerned.

4.7 The methods based on the measurements of unsteady fluctuating pressures on rigid models give accurate predictions as shown through comparison with revised calculations of the measured response on windtunnel models and through comparison with flight test data. The prediction may be poor if the method uses 100 % aerodynamic damping from linear unsteady theory for all modes, especially for the wing second bending and first torsion modes.

4.8 There is strong response in the buffet penetration region in the first wing torsion mode resulting sometimes in torsional buzz, and in the second wing bending mode. This is partly due to shock wave oscillation. The various phenomena have been observed in aeroelastic model testing and flight testing and deserve special attention.

4.9 Dynamic tail load estimations from direct measurements on rigid models are open to question because of significant flexible mode response and non-simulation of modes and frequencies. The need for accurate prediction of tail loads particularly for highly swept wing - tail combination increases. Tail buffet loads may be more critical than wing buffet loads.

4.10 The magnitude of the mean velocity is not a good inicator of jet engine and spoiler wake intermittant flow regions which will be much nearer the tail than indicated by connventional pitot rake measurements. Larger tail buffet responses may occur than indicated by such simple measurements.

4.11 Propeller wake excitations cause high, discrete frequency responses of tail components. The loadings cannot be predicted early with accuracy. Maximum tail excitation occurs when the flow is parallel to the horizontal tail. Fatigue failures are a distinct possibility and are avoided by a careful flight and ground measurement program and evaluations.

4.12 Leading edge vortex effects at low angles of attack appear to cause noticeable drops in pretícted flutter speeds. Additional experimental and analytical studies are needed.

5. RECOMMENDATIONS

Some general suggestions are as follows:

- Correction methods for existing lifting surface theories for prediction of unsteady subsonic and supersonic flow and airloads.
- Introduction of shock wave oscillation effects and unsteady flow pattern through introduction of measured fluctuating pressures on wing upper surface regions and calculation of aerodynamic damping or excitations in different modes. Calculation of the influence of these unsteady pressures on attached flow region unsteady forces. The same procedure is recommended for wing horizontal tail interactions to predict additional fluctuating damping or excitation due to separated wing flow in the subsonic and supersonic case.
- Increase emphasis on the prediction of the buffet response on tails. Prediction methods for tail buffet will have to account for wing wake position.
- Further examination of the correction procedure through comparison test and calculation.
- Further investigations of Reynolds number effects on separated flow pattern on wing upper side.
- Investigation of vibration amplitude effects on separated flow patterns and intensity of fluctuating pressures by the use of harmonically oscillating rigid models of different materials.

- Further investigation of the strong torsional and second bending responses by various aeroelastic modeling approaches.
- Further measurement of structural and aerodynamic damping in model and flight testing. Correlation of measured aerodynamic damping with analytical predictions.
- Detailed buffet measurements on wings with stores.

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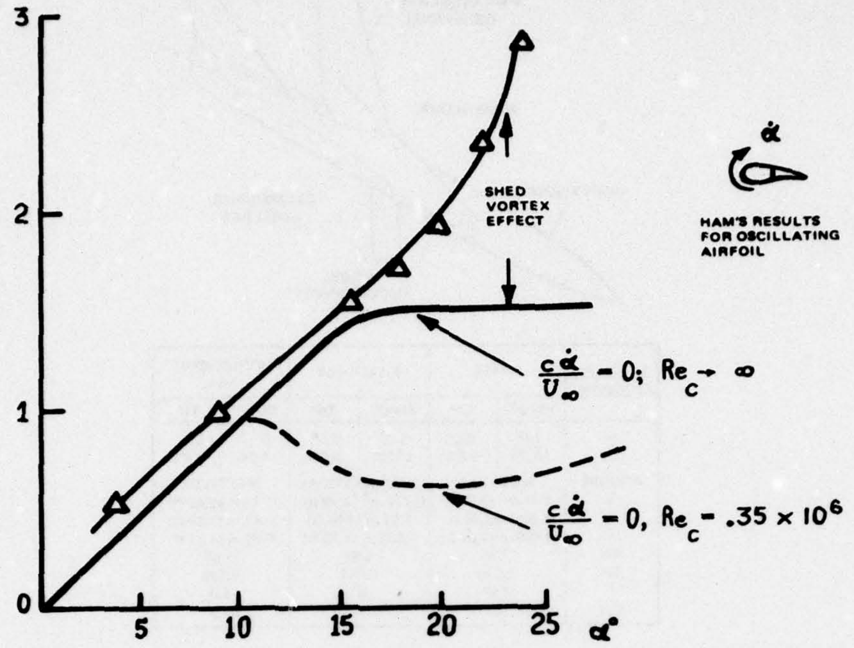


Fig.1 Nonlinear vortex – induced lift

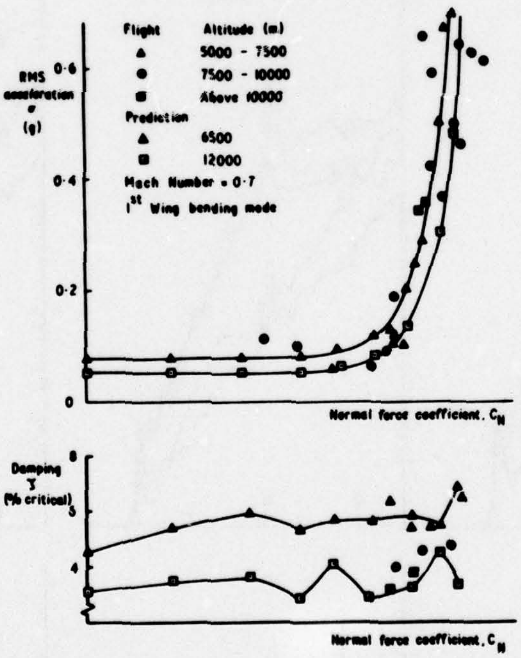
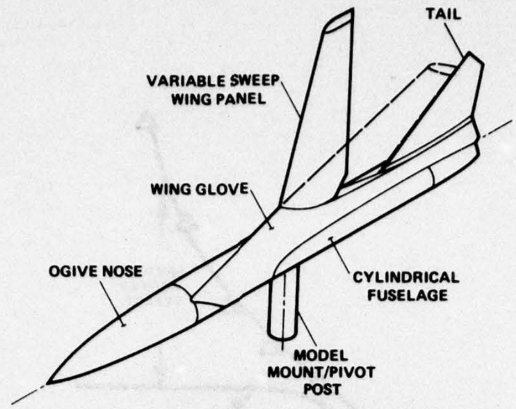


Fig.2 Predicted and measured rms wing tip acceleration, σ and damping τ v normal force coefficient, C_N : M = 0.7



SEMISPAN GEOMETRY	F-111A		F-111 TACT		HORIZONTAL TAIL	
	PIVOT	TIP	PIVOT	TIP	ROOT	TIP
i	1.0°	-3.0°	1.0°	-6.5°	0°	0°
t/c	10.7%	9.8%	10.2%	5.4%	4.0%	3.0%
AIRFOIL	NACA 64A		SUPERCRITICAL		BICONVEX	
S	$0.677\ m^2$ ($7.29\ ft^2$)		$0.779\ m^2$ ($8.39\ ft^2$)		$0.23\ m^2$ ($2.42\ ft^2$)	
b	1.60 m (5.25 ft)		1.51 m (4.94 ft)		0.42 m (1.36 ft)	
$\bar{c}$	0.460 m (1.51 ft)		0.532 m (1.75 ft)		0.58 m (1.91 ft)	
AR	7.56		5.83		1.54	
TR	0.325		0.541		0.334	
Γ	1.0°		0°		-1.0°	
Λ_{LE}	16° to 72°		16° to 58°		57.5°	

Fig.3 Geometry of 1/6-scale semispan F111A and TACT models

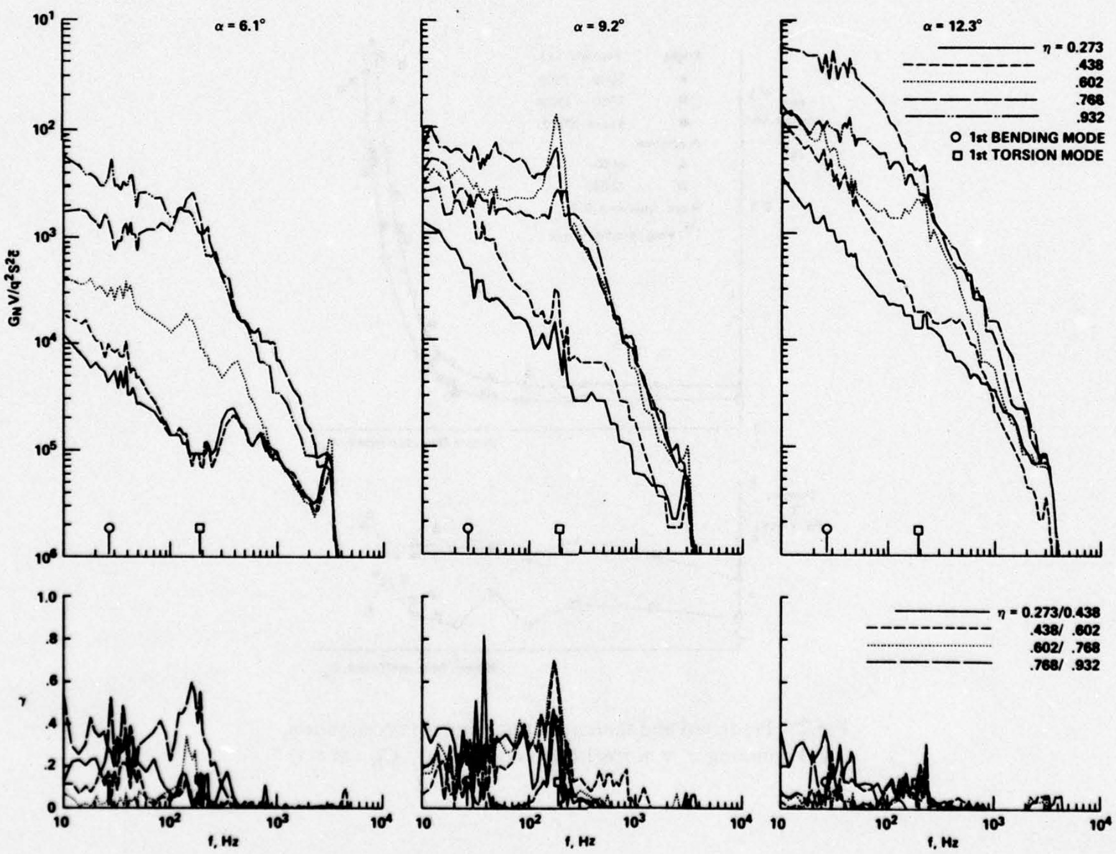


Fig.4 Power spectra and coherence of section normal-force fluctuations on 1/6-scale F-111A model for $\Lambda = 26°$, $M = 0.85$

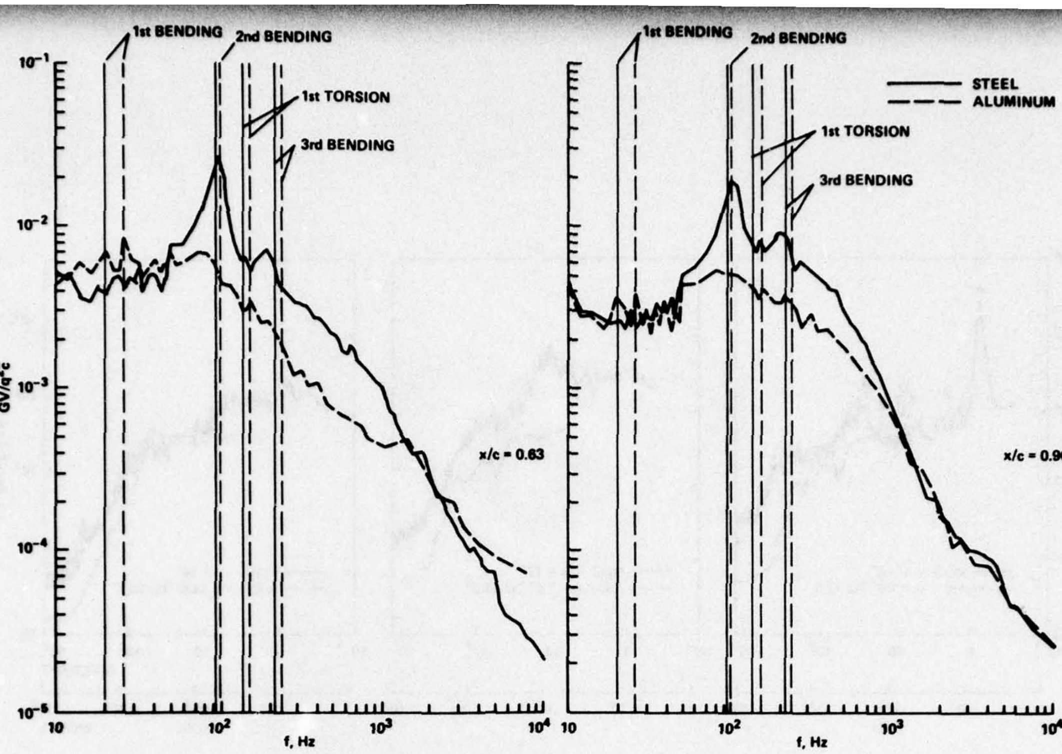


Fig.5 Power spectra of pressure fluctuations on 1/6-scale TACT models at $\eta = 0.744$ for $\Lambda = 26°$, $R = 10.5 \times 10^6$, $M = 0.90$, $\alpha_P = 12°$

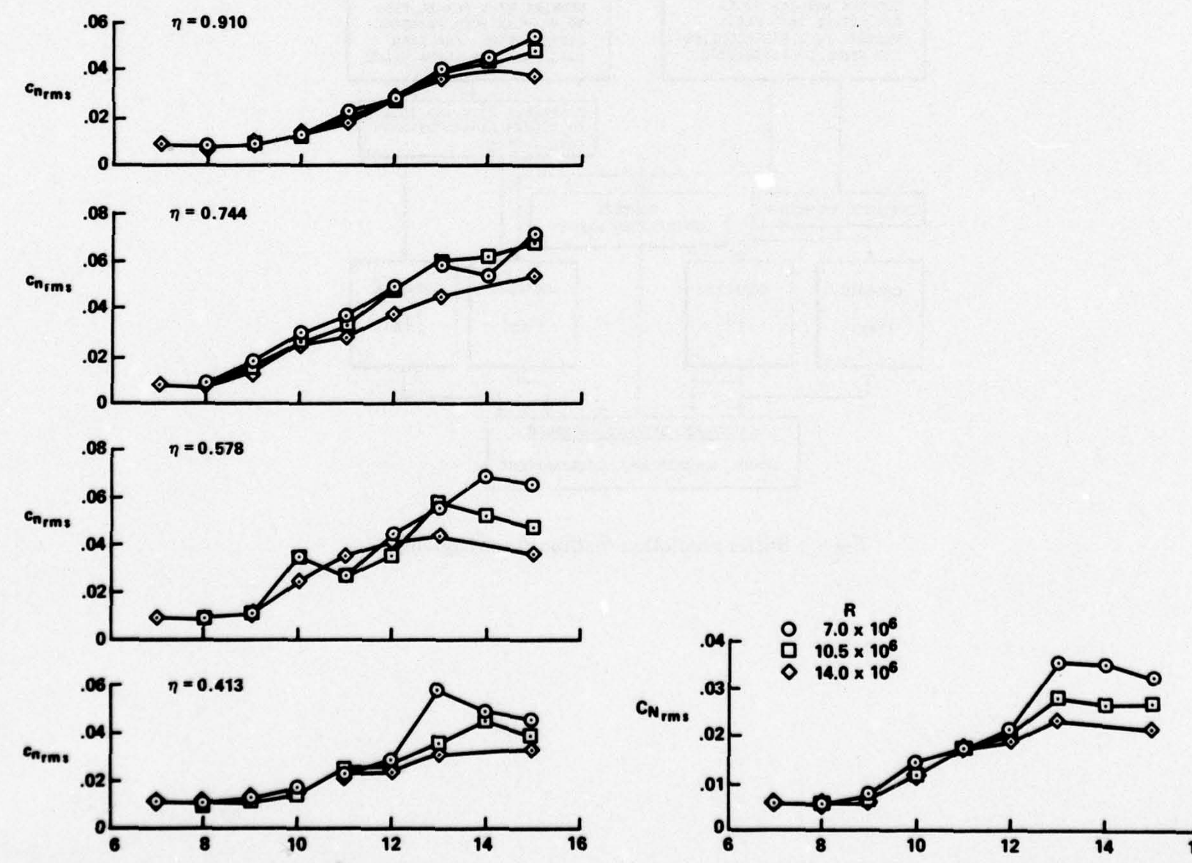


Fig.6 Fluctuations of section and total normal force on 1/6-scale steel TACT model for various Reynolds numbers at $\Lambda = 26°$, $M = 0.80$

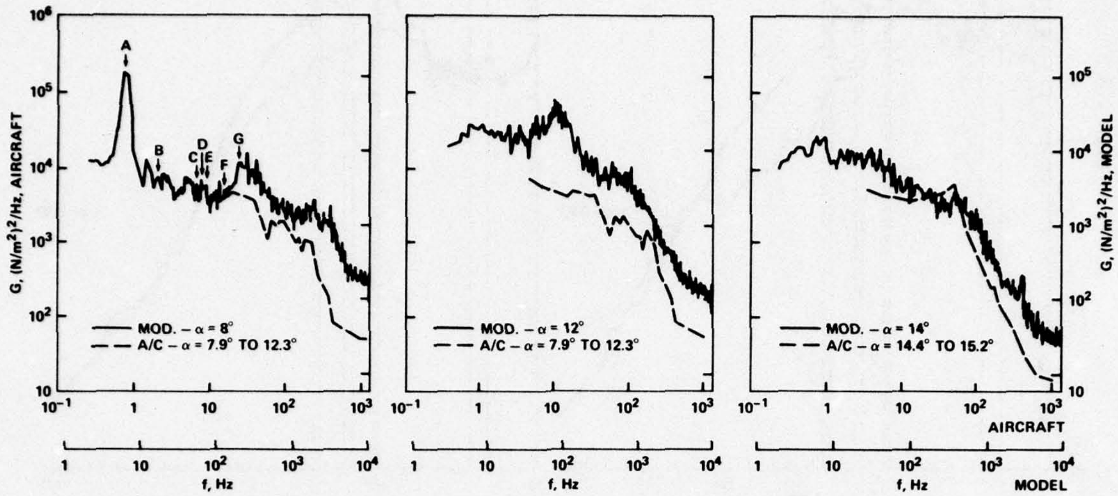


Fig.7 Comparison of power spectra of pressure fluctuations on F-5A 1/7-scale model and aircraft from transducer 11 at $M = 0.75$, $R_{mod} = 4.71 \times 10^6$, $R_{flt} = 18.9 \times 10^6$

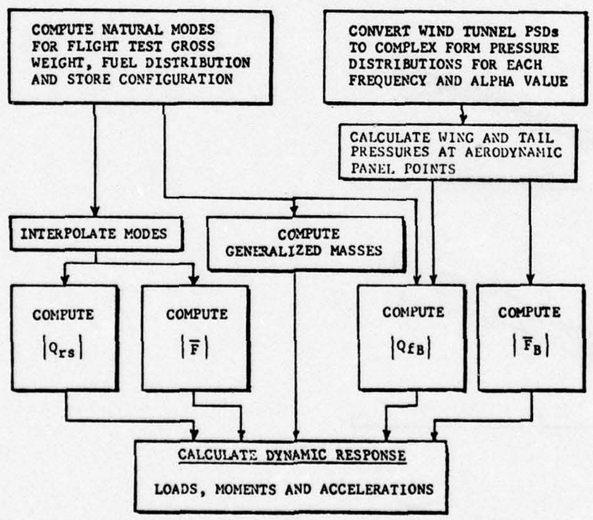


Fig.8 Buffet prediction method flow diagram

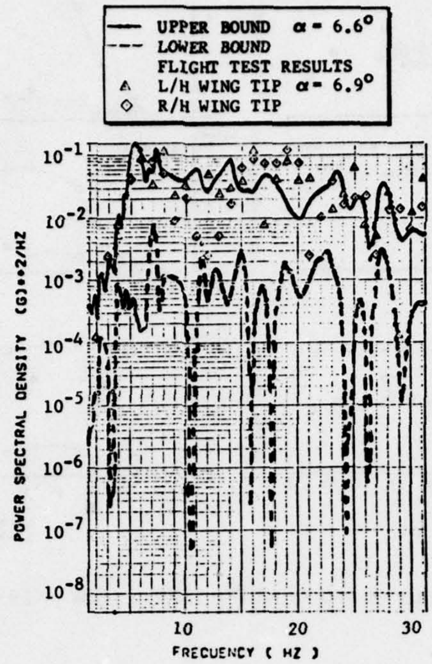


Fig.9 Predicted bounds on wing tip accelerometer PSD for $\Lambda = 26°$, M = 0.8, ALT = 6035 m

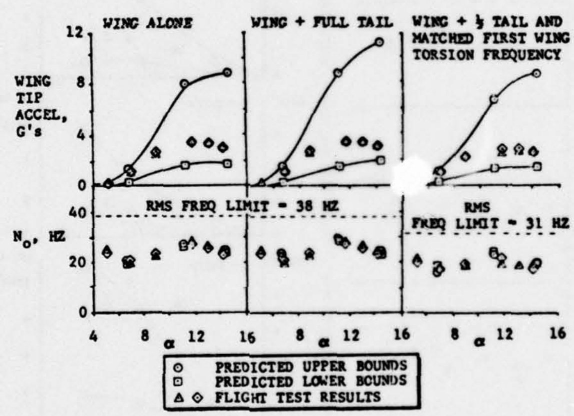


Fig.10 Effect of horizontal tail loads on wing tip accelerometer for $\Lambda = 26°$, M = 0.8, ALT = 6035 m

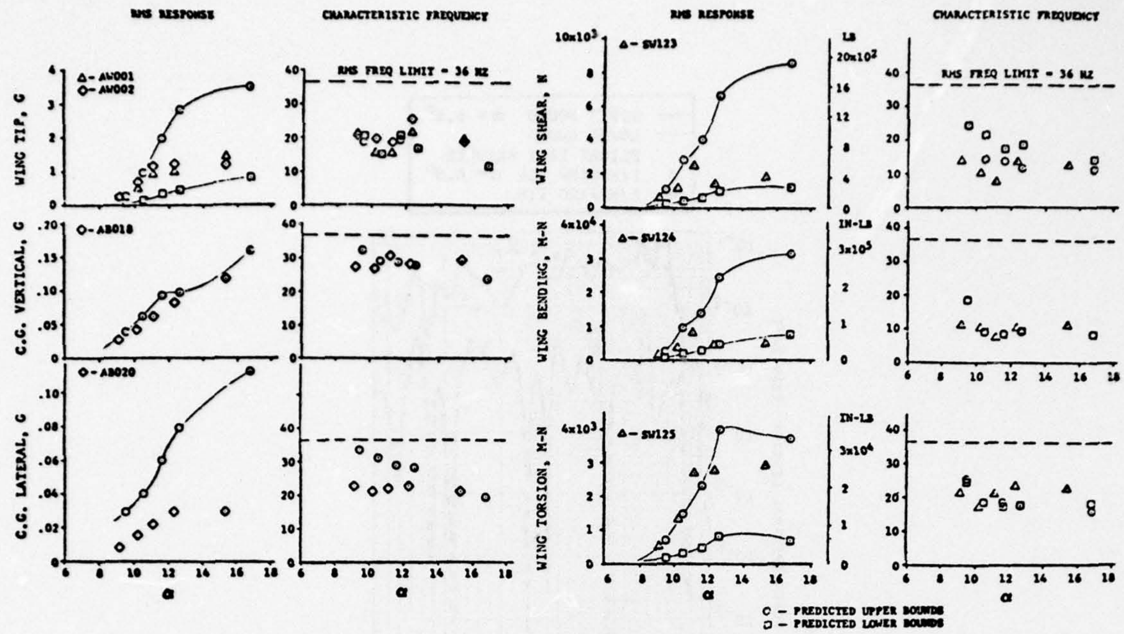


Fig.11 Buffet response for $M = 0.7$, $ALT = 7559$ m, G.W. = 293138N, and $\Lambda = 26°$

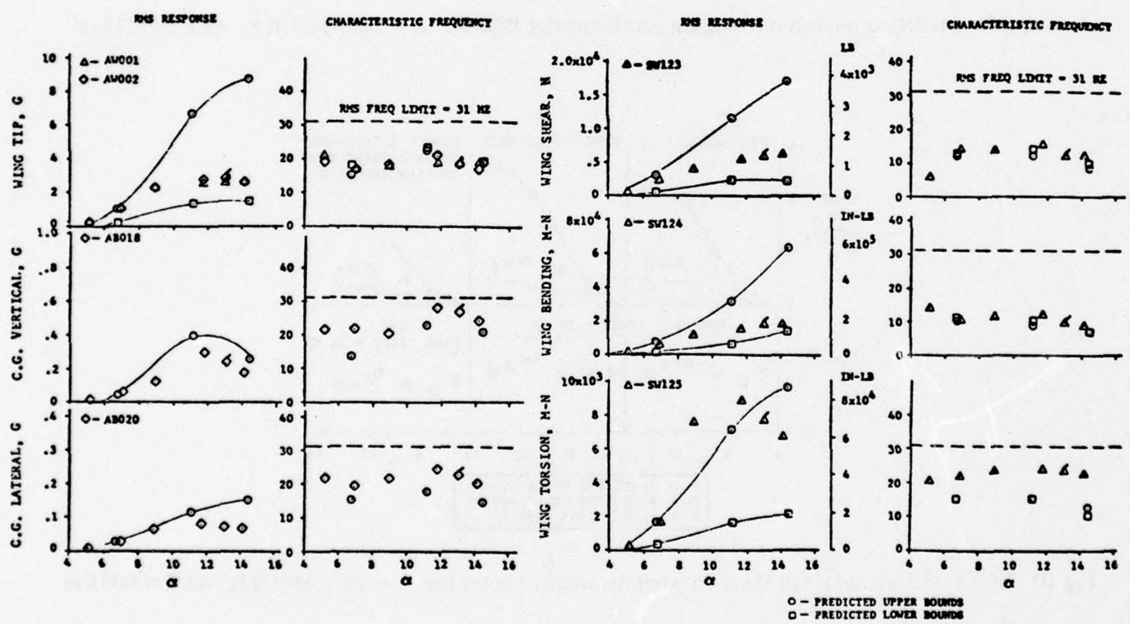


Fig.12 Buffet response for $M = 0.8$, $ALT = 6035$ m, G.W. = 266448N, and $\Lambda = 26°$

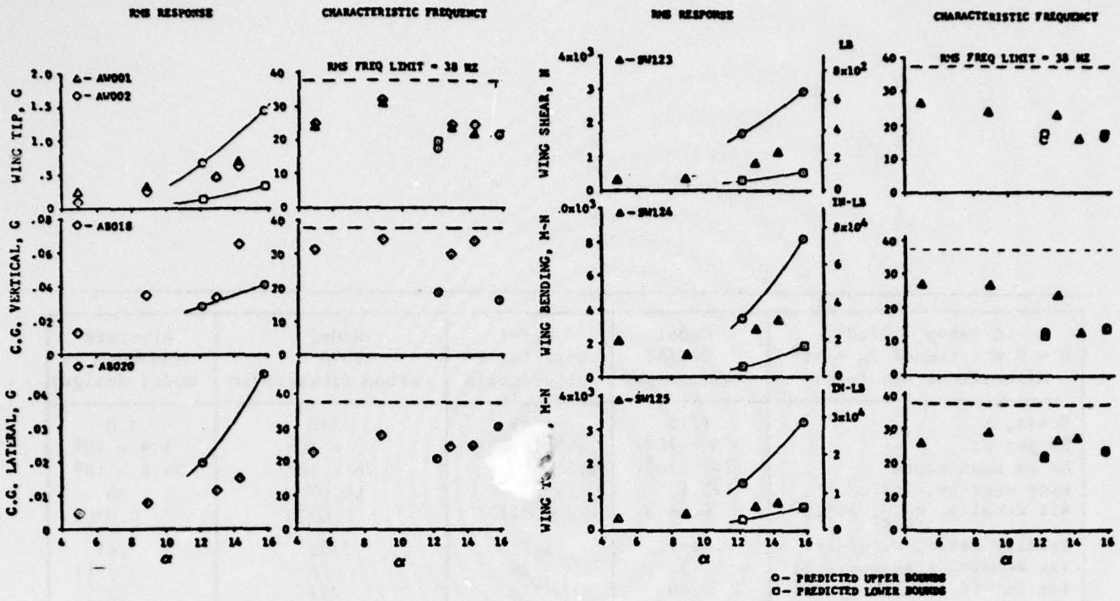


Fig.13 Buffet response for M = 1.2, ALT = 9053 m, G.W. = 261778N, and Λ = 50°

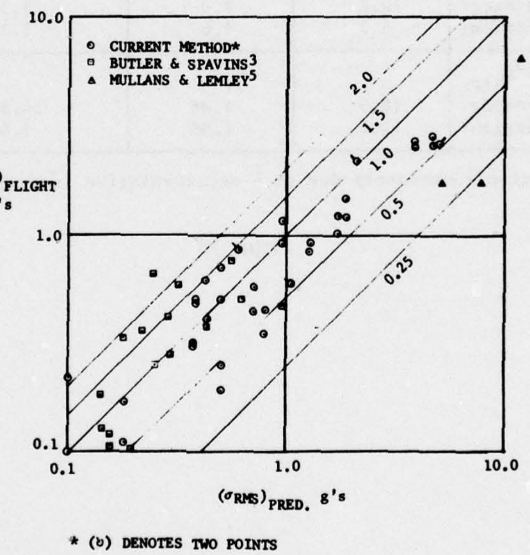


Fig.14 Correlation of prediction and flight test data for wing tip accelerometer

LE sweep : 27.2° M = 0.8 : tunnel T_0 = 45° Aircraft at sea level	Model 577/Al solid metal	Model 577/Flex 2 steel/resin	Model 2070 carbon fibre/resin	Aircraft (assumed in model design)
Scale, S	1/15	1/15	1/10	1.0
RN per ft	3×10^6	3×10^6	2×10^6	5.9×10^6
RN on mean chord	1.26×10^6	1.26×10^6	1.26×10^6	24.6×10^6
Wing density, lb/ft^3	173.1	143.4	66.02	20
Air density, ρ_{air}, lb/ft^3	0.0413	0.0413	0.0275	0.0766
Density ratio, wing/air	4191	3472	2383	261
1st bending frequency, Hz	77	39	75	(7½)
1st torsion frequency, Hz	760	275	287	(30)
Relative frequency × S				
1st bending:	0.68	0.35	1.0	1.0
1st torsion:	1.69	0.61	0.96	1.0
Relative mass/S^3	8.6	7.2	3.3	1.0
*Relative stiffness/S^3				
bending:	10.2	1.0	5.33	1.0
torsion:	8.2	1.0	1.32	1.0
*Relative stiffness/S$^3\rho_{air}$				
bending:	18.9	1.85	14.8	1.0
torsion:	15.2	1.85	3.65	1.0

* Based on the deflection root-to-tip due to a representative loading, summed across the span.

Figure 15

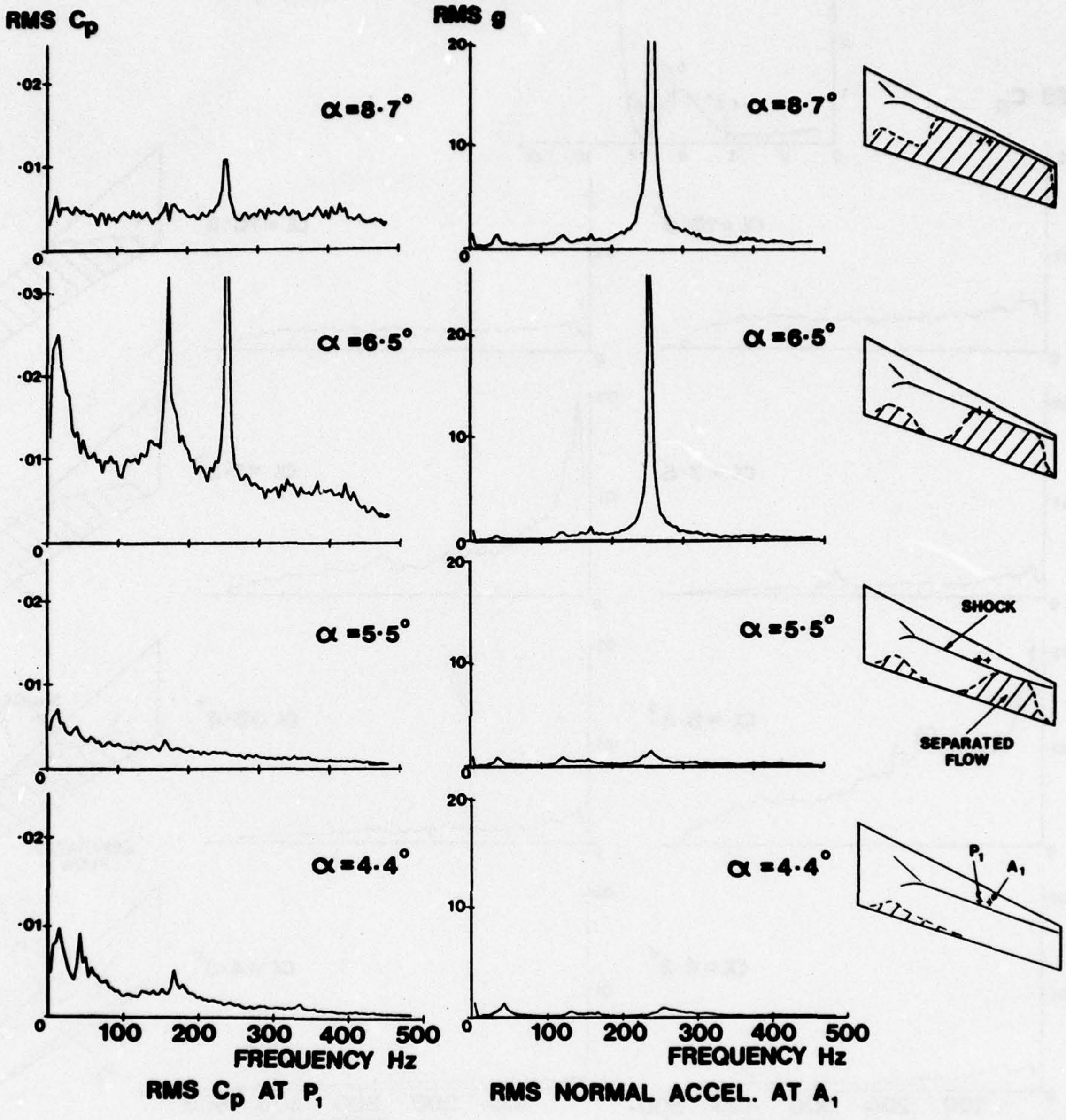


Fig.16 Unsteady aerodynamic pressure and structural response at wing-torsional buzz conditions. Model 577/Flex 2, leading-edge sweep 27.2°, M = 0.75, $R_{\bar{c}} = 1.26 \times 10^6$

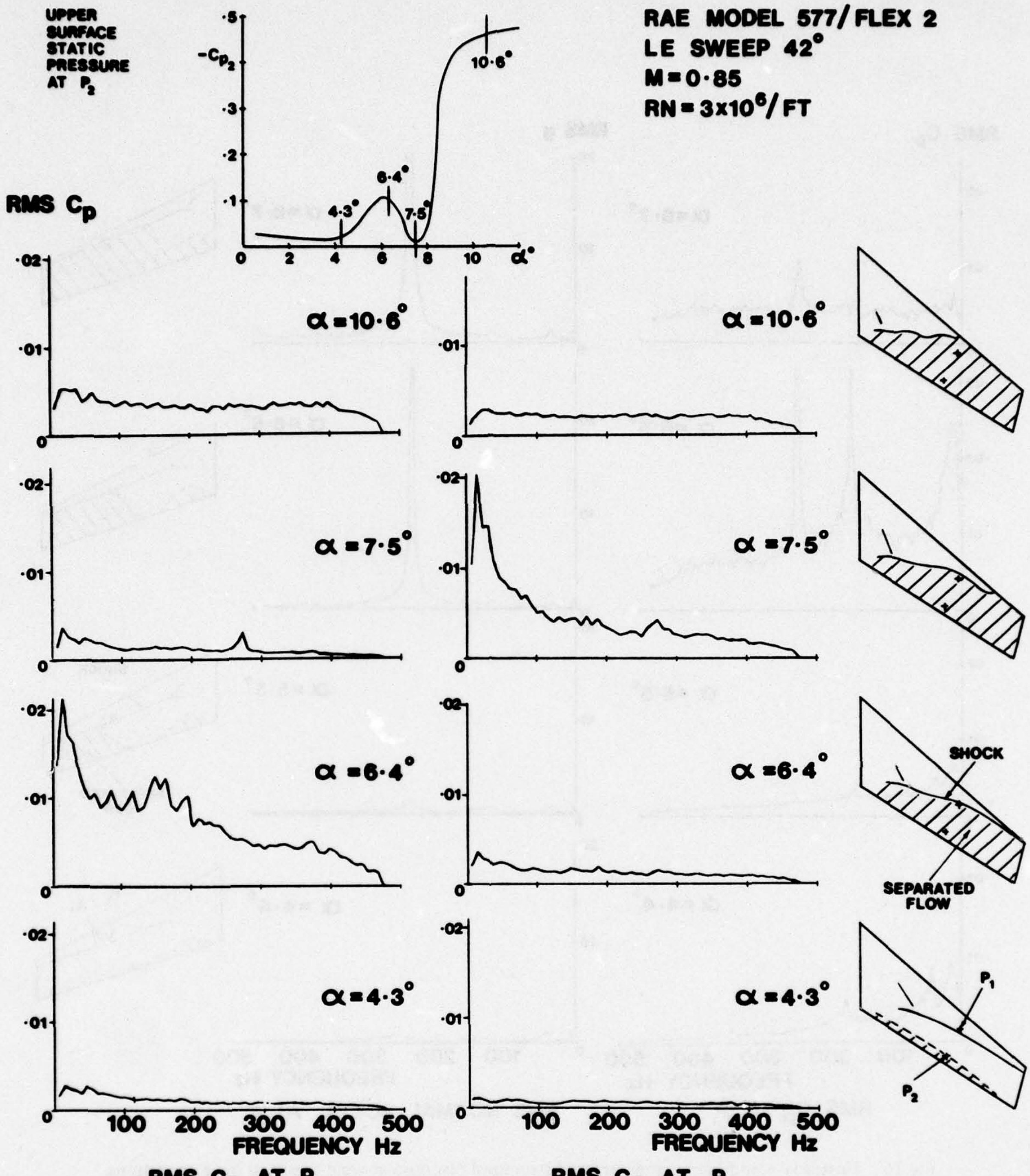


Fig.17 Unsteady aerodynamic pressures on a swept wing at buffeting conditions

FREQUENCY PARAMETER OF PRIMARY TORSION MODE, $\omega\bar{c}/V$

0.87	1.36	2.40

FREQUENCY OF PRIMARY TORSION MODE RELATIVE TO NOMINAL AIRCRAFT

0.61	0.96	1.69

TORSIONAL STIFFNESS RELATIVE TO NOMINAL AIRCRAFT

1.0	1.32	8.2

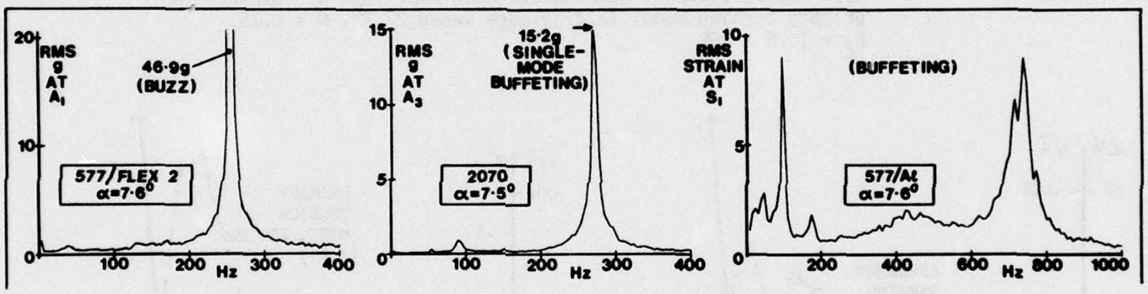
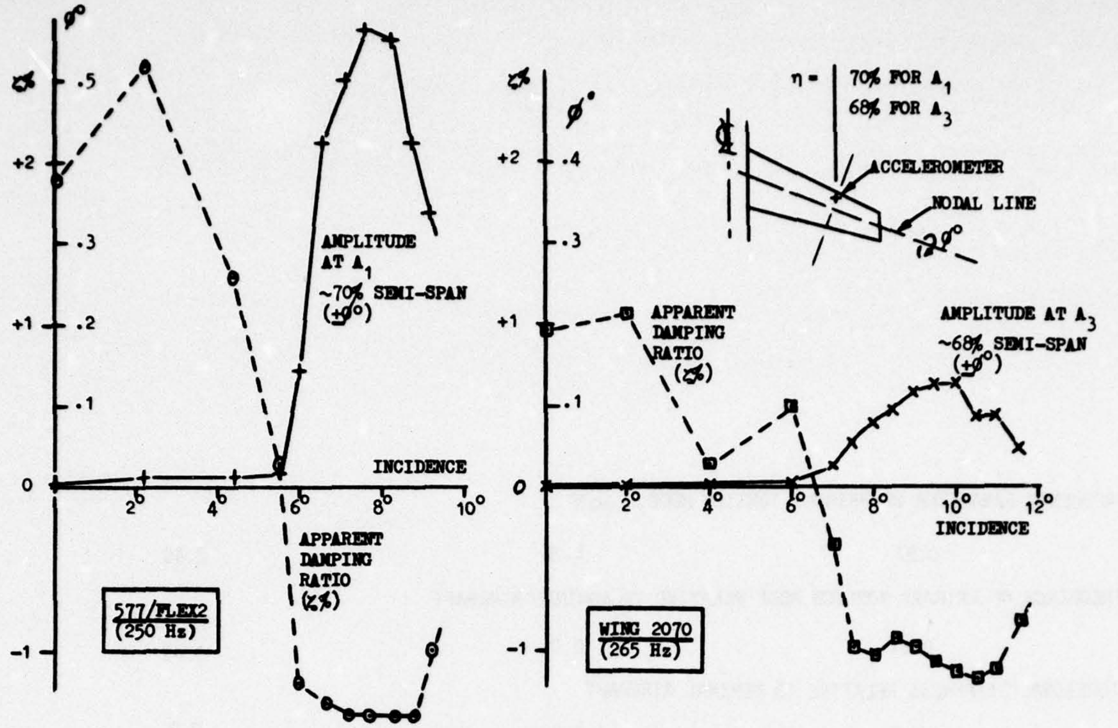


Fig.18 Structural response of the three wings of Figure 8 at a high incidence; leading-edge sweep 27.2°, M = 0.75, $R_{\bar{c}} = 1.26 \times 10^6$



Apparent aerodynamic damping and mean amplitude of response in primary torsion mode: leading-edge sweep 27.2°, M = 0.75, $R_{\bar{c}}$ = 1.26 x 10^6

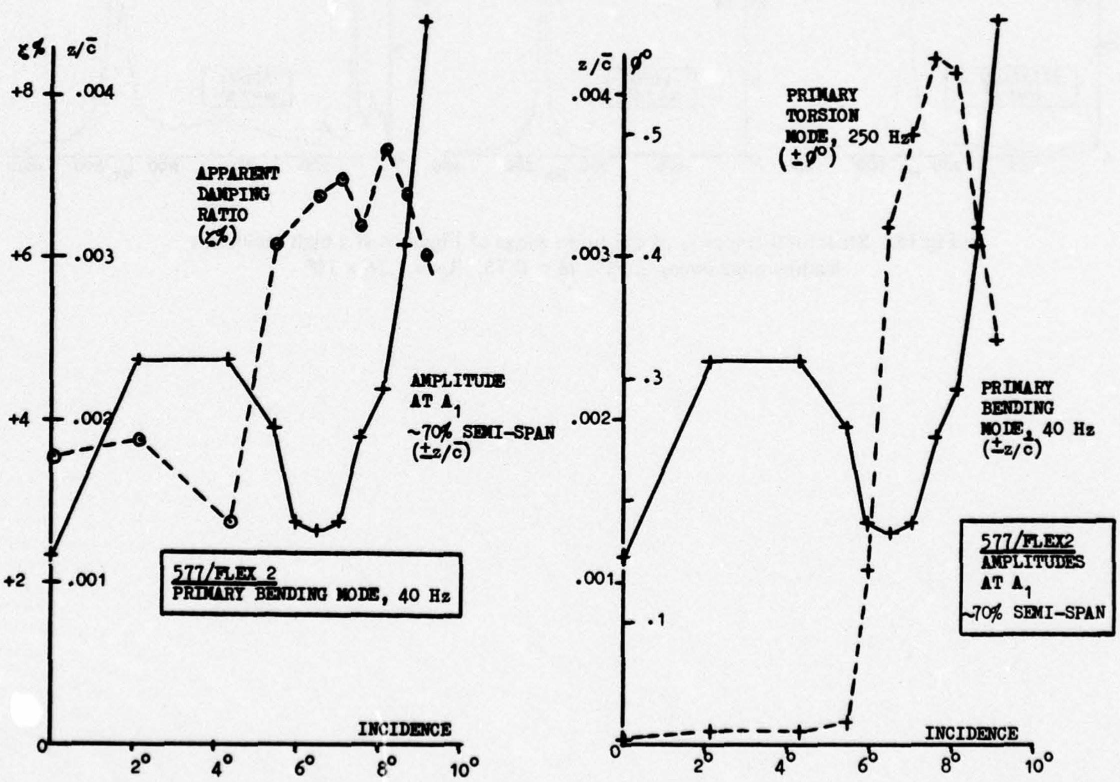


Fig.19 Apparent aerodynamic damping and mean amplitude of response of 577/Flex 2 wing: leading-edge sweep 27.2°, M = 0.75, $R_{\bar{c}} = 1.26 \times 10^6$

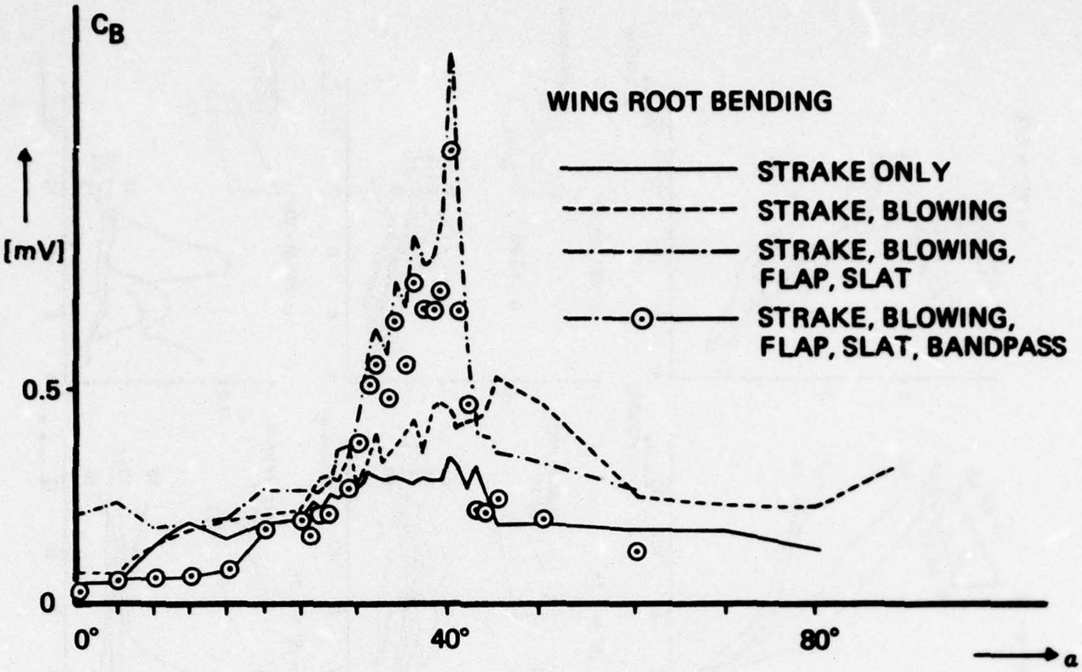


Fig.20 Filtered and not filtered rms wing root bending moment of the different conditions

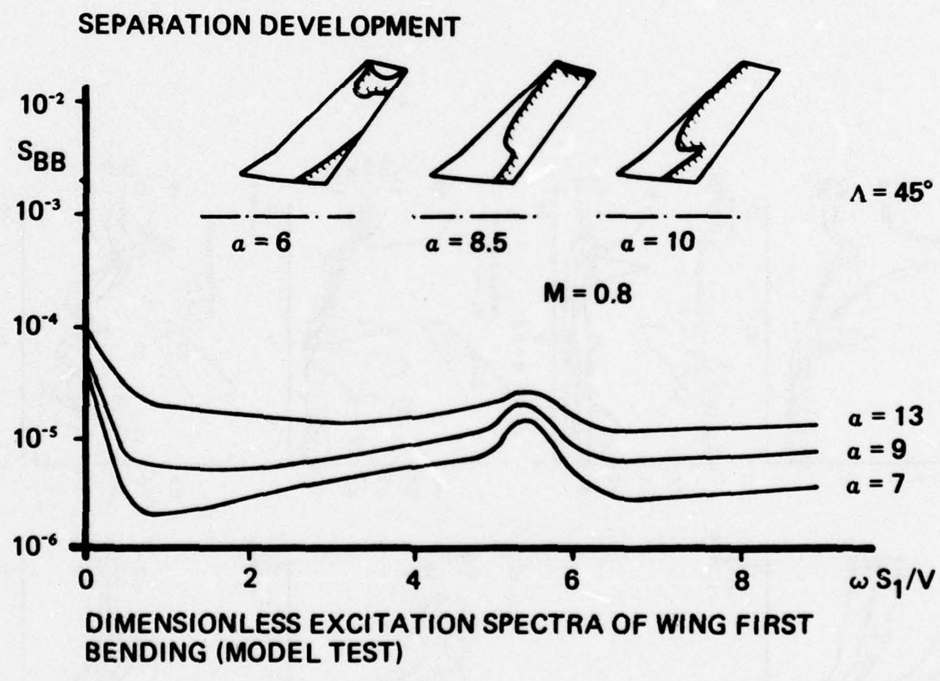


Fig.21 Dimensionless excitation spectra of the true aircraft at $M = 0.7$, $\Lambda = 25°$ and $M = 0.8$, $\Lambda = 45°$

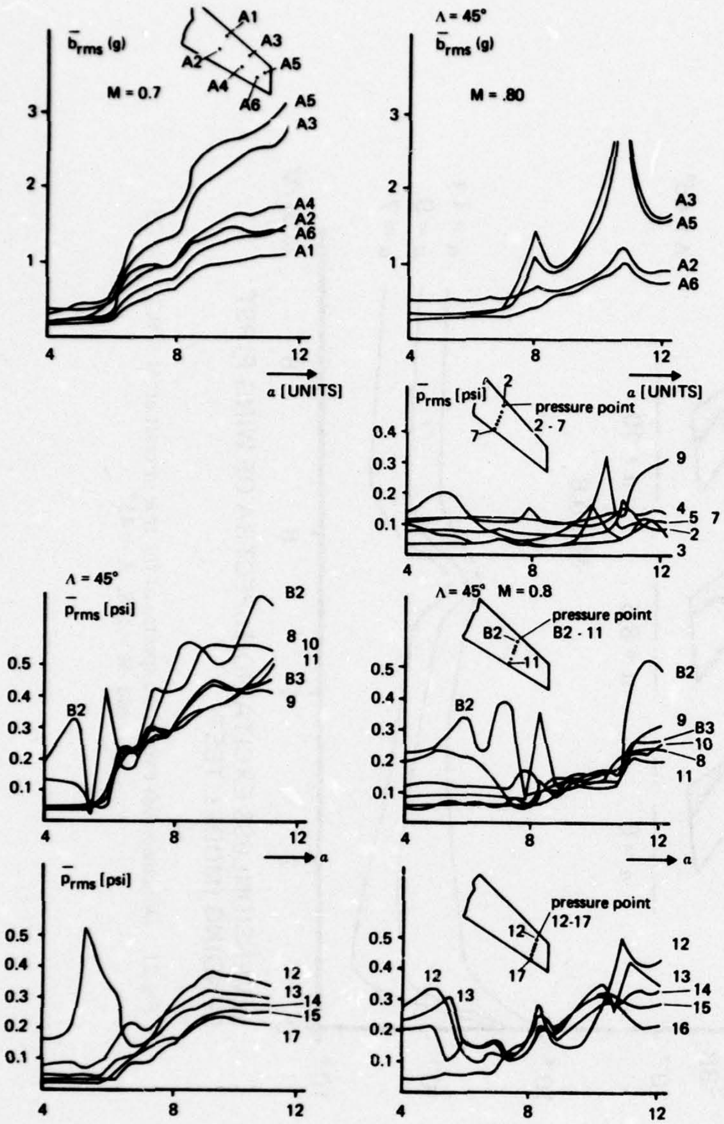


Fig.22a Influence of incidence and Mach number on the rms pressures and accelerations on the wing at $\Lambda = 45°$

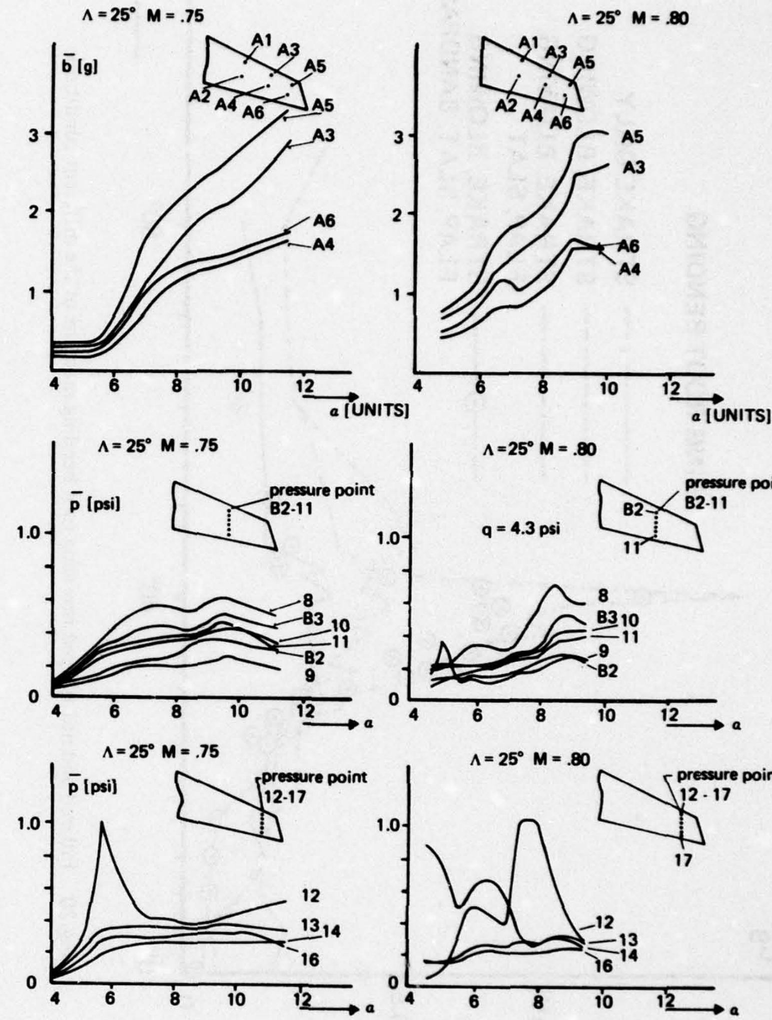


Fig.22b Influence of incidence and Mach number on the rms pressures and accelerations on the wing at $\Lambda = 25°$

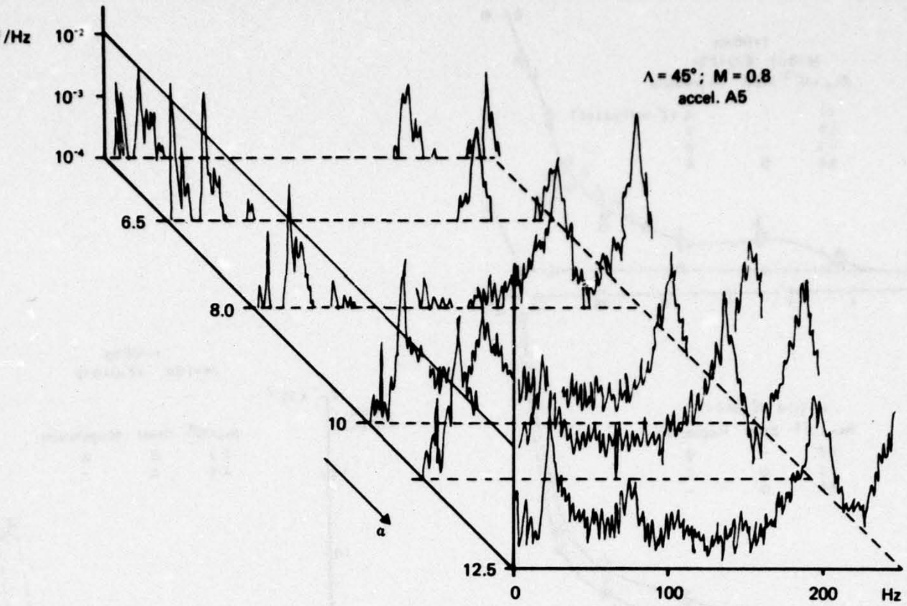
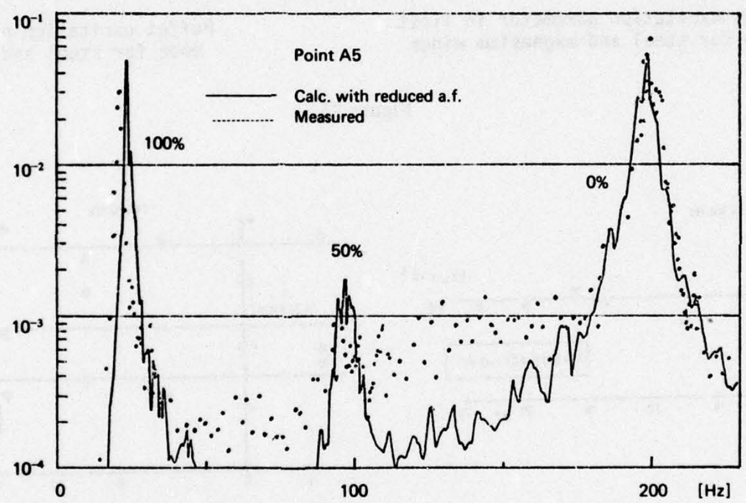


Fig.23 Influence of incidence on the acceleration spectra at $\Lambda = 45^o$, M = 0.8



PSD of wing tip acceleration, $\alpha = 12^o$, M = 0.7, $\Lambda = 45^o$

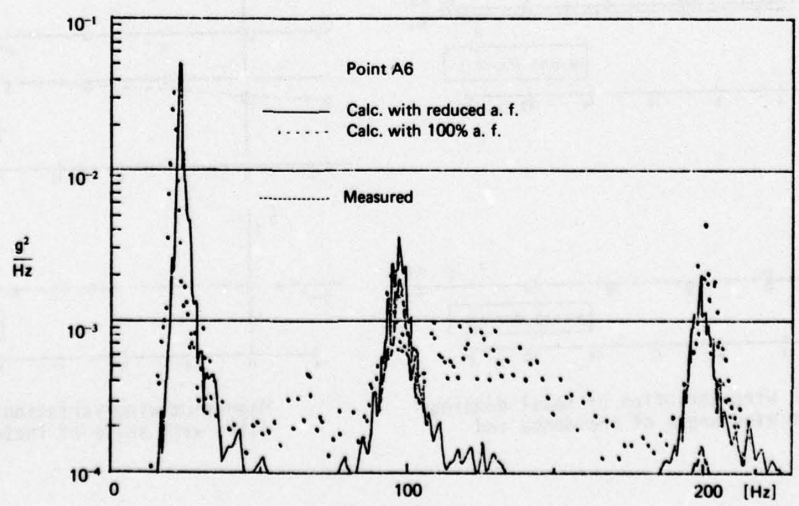
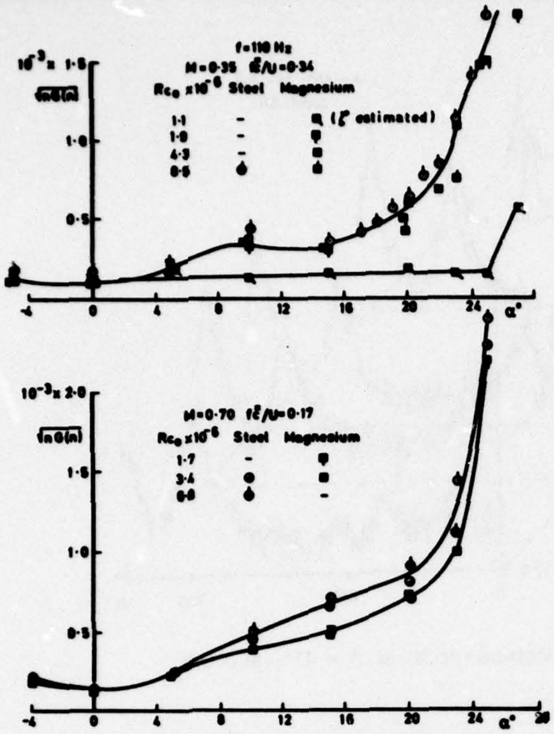
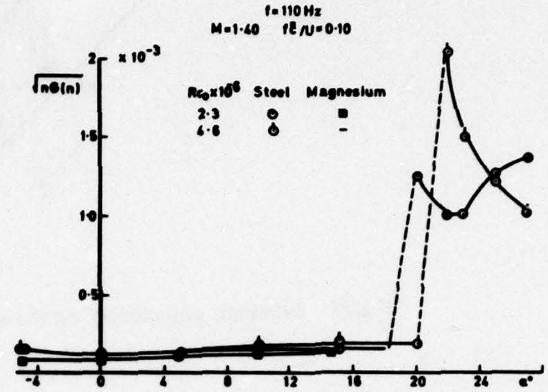


Figure 24

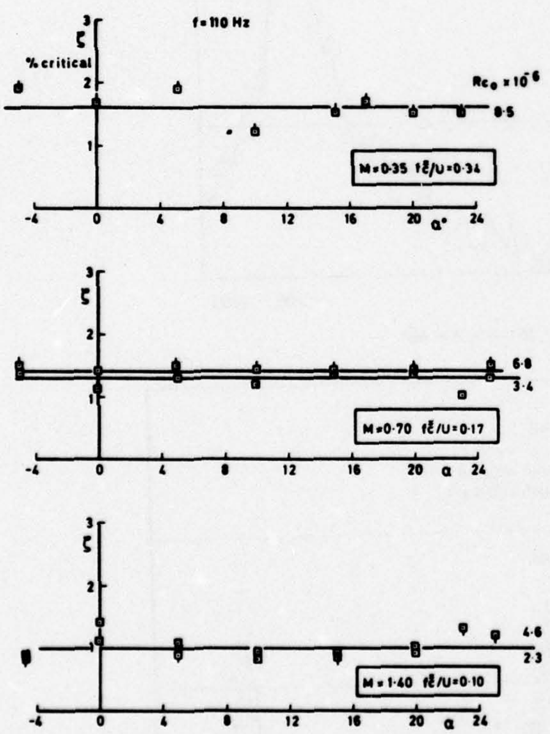


Buffet excitation parameter in first mode for steel and magnesium wings

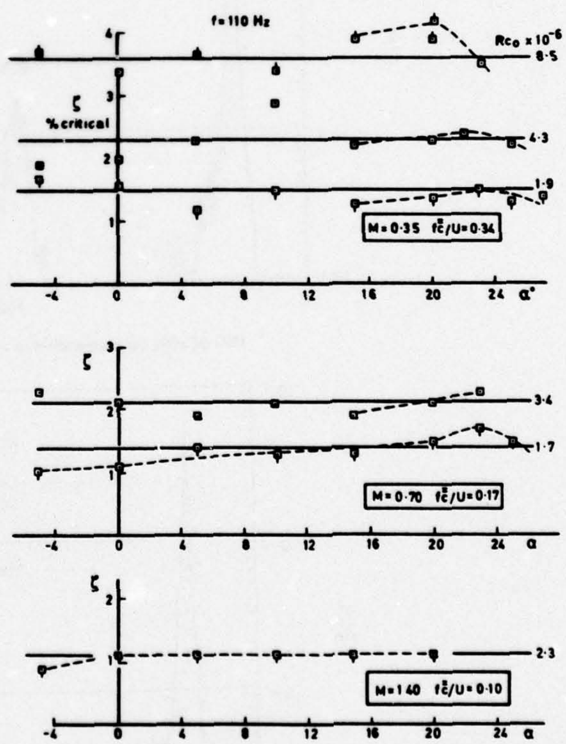


Buffet excitation parameter in first mode for steel and magnesium wings

Figure 25



Steel wing-variation of total damping ratio wing angle of incidence and



Magnesium wing-variation of total damping ratio with angle of incidence and Mach

Figure 26

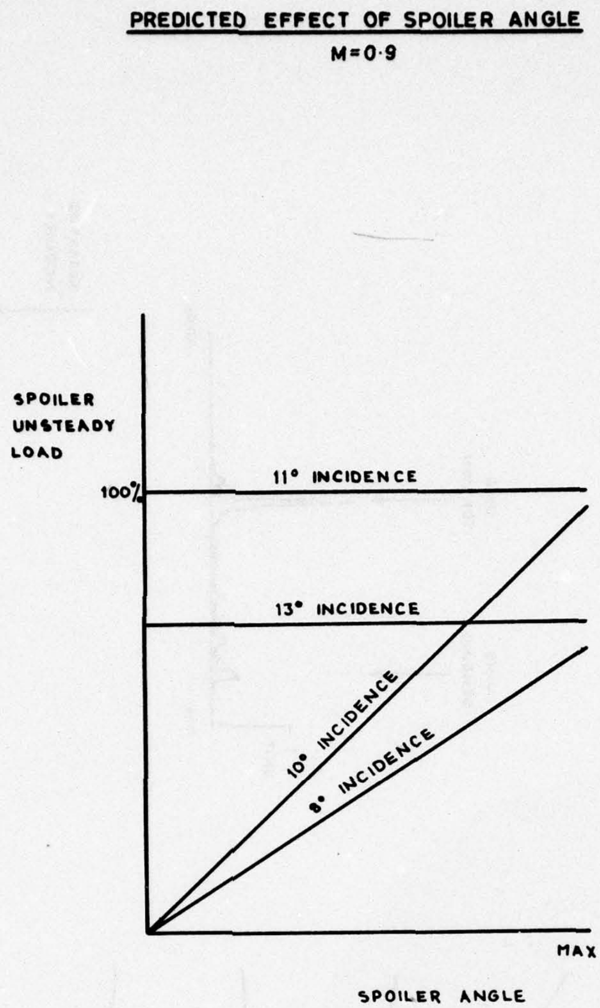


Figure 27

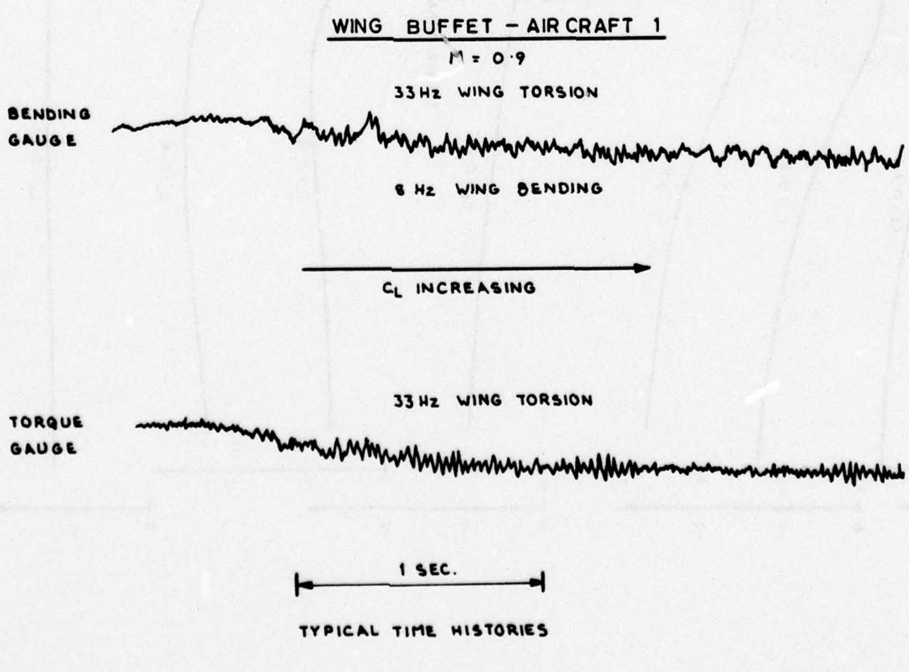


Figure 28

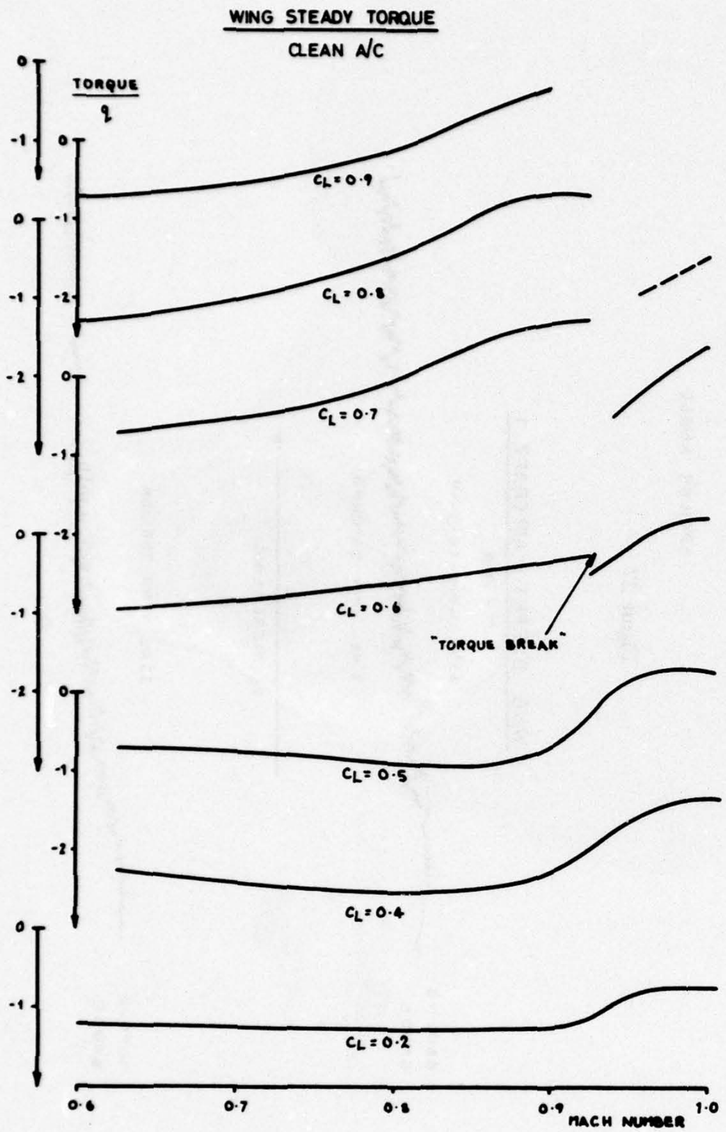


Figure 29

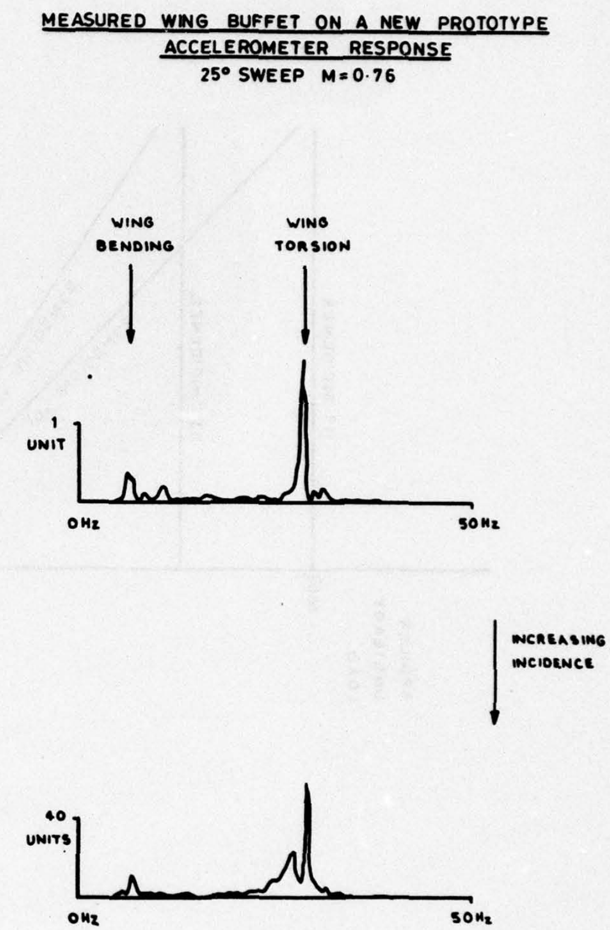


Figure 30

TYPICAL VARIATION OF TAIL ROOT BENDING MOMENT COEFFICIENT

$C_{TRB} = \frac{\text{TAILERON ATTACHMENT LOAD}}{\frac{1}{2}\rho V^2}$

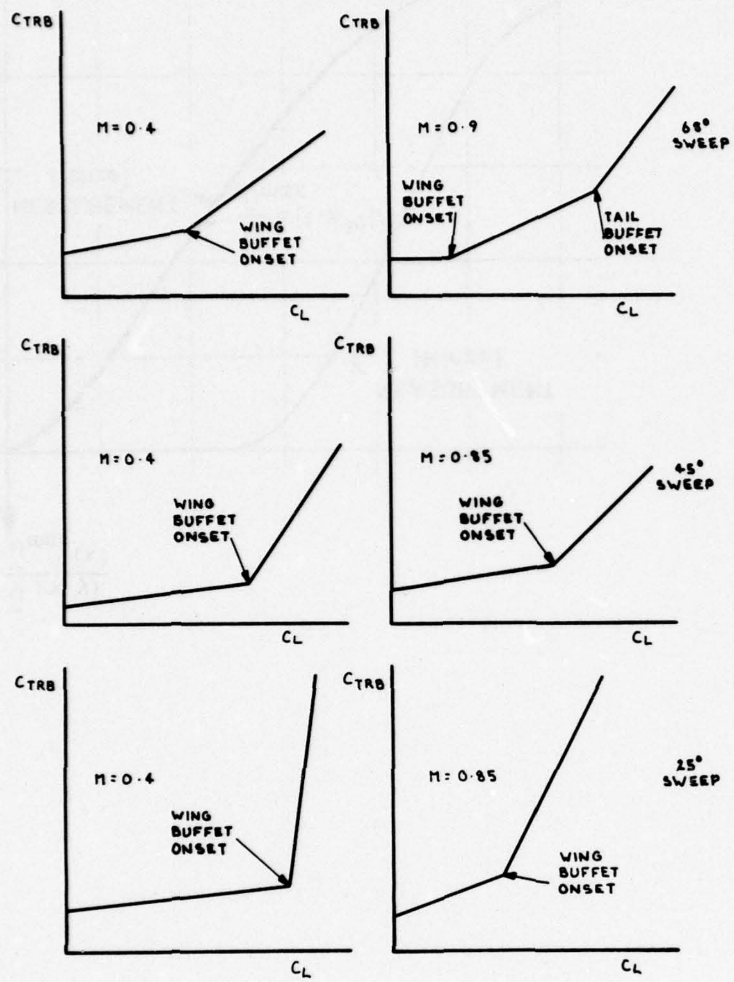
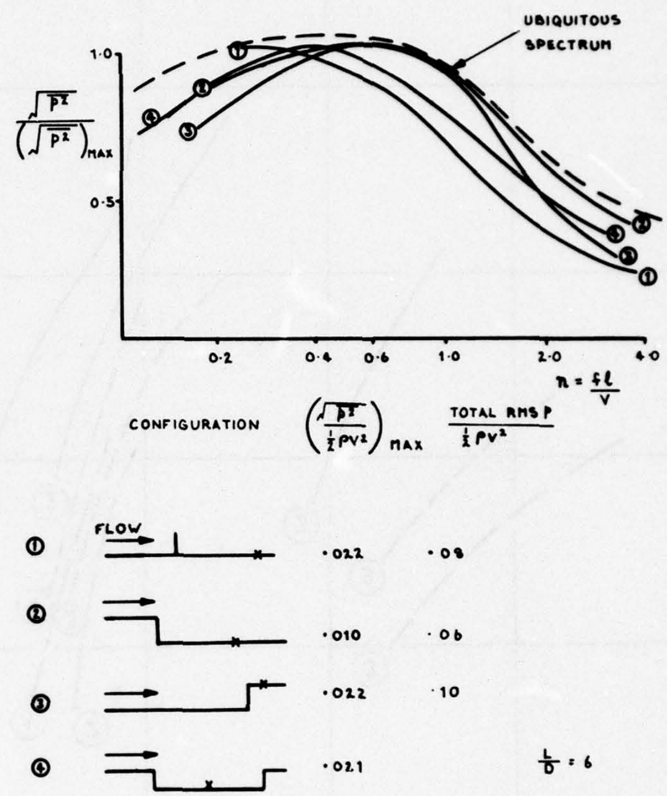


Figure 31

RANDOM PRESSURE FLUCTUATIONS IN SEPARATED FLOW REGIONS - SUBSONIC FREE STREAM

(REF 2)



l IS SCALE OF SEPARATED FLOW REGION

$\sqrt{\overline{p^2}}$ IS RMS PRESSURE IN BANDWIDTH 0·1n

x MEASUREMENT POINT

Figure 32

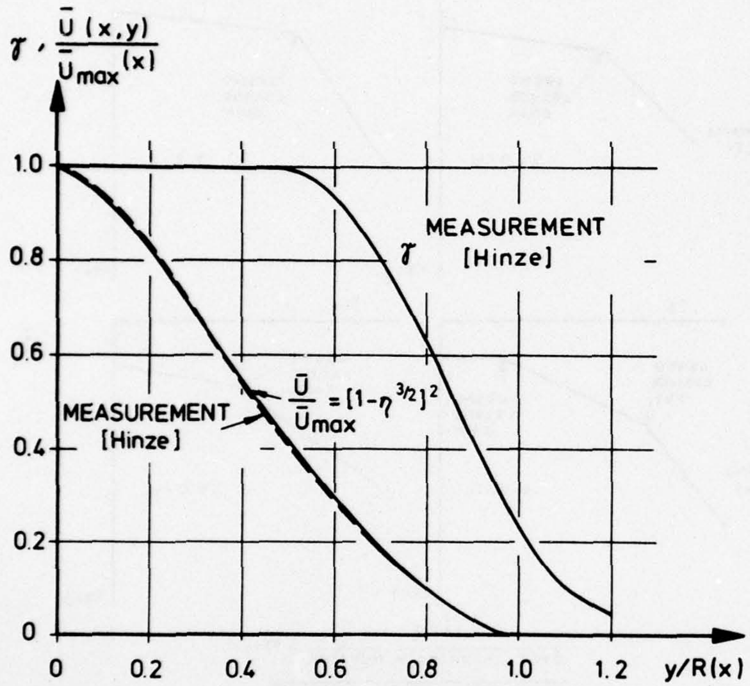


Fig.33 Intermittency factor and velocity profile

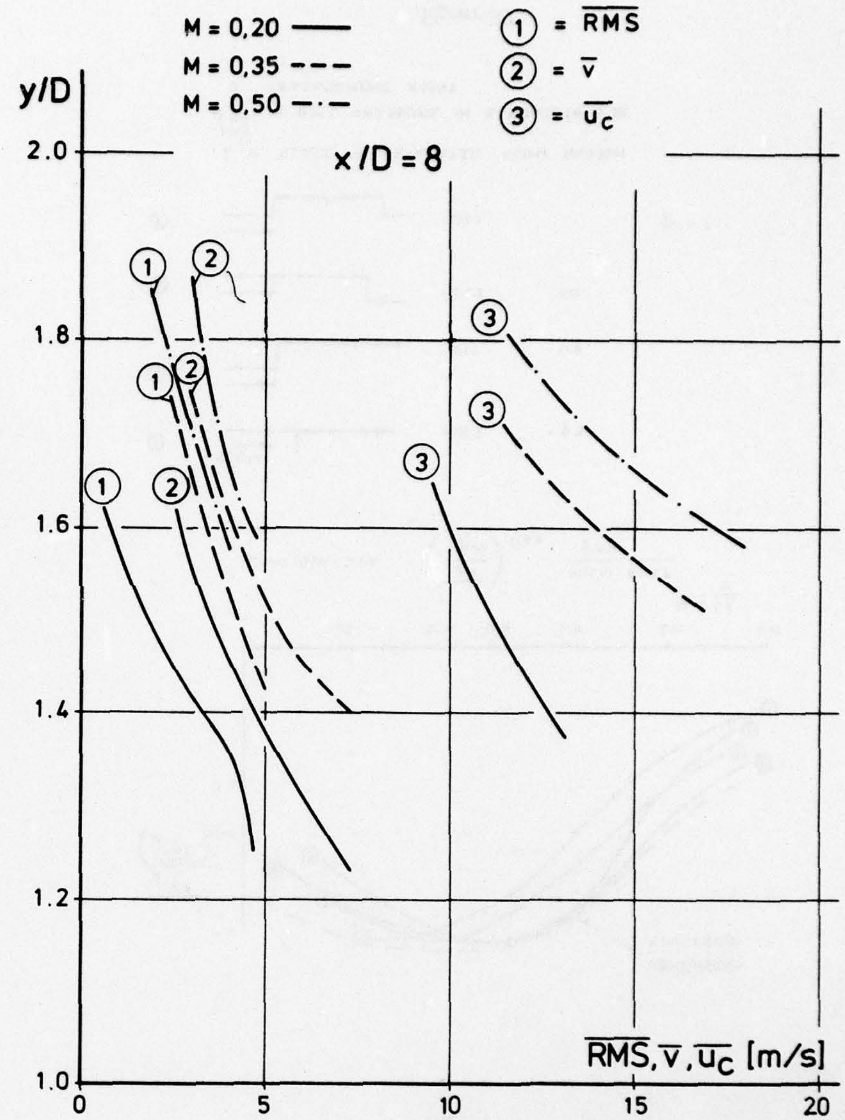


Fig.34 Velocity fluctuations in a circular jet

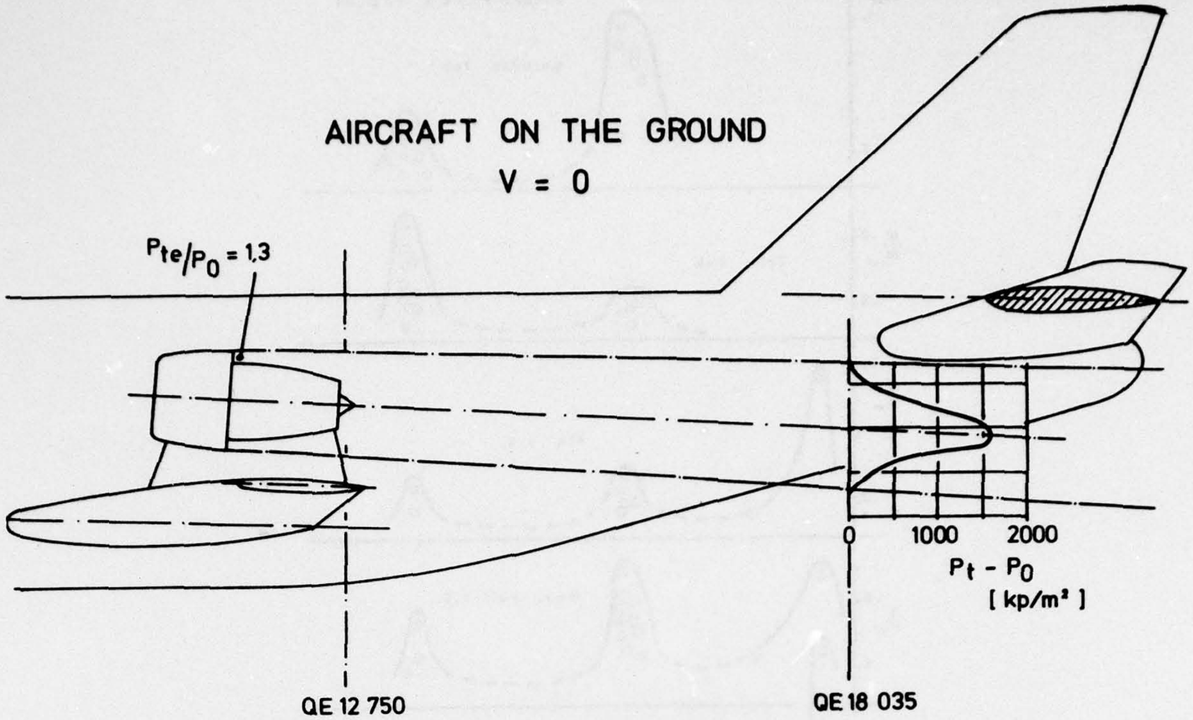


Fig.35 Engine jet spreading underneath the horizontal tail

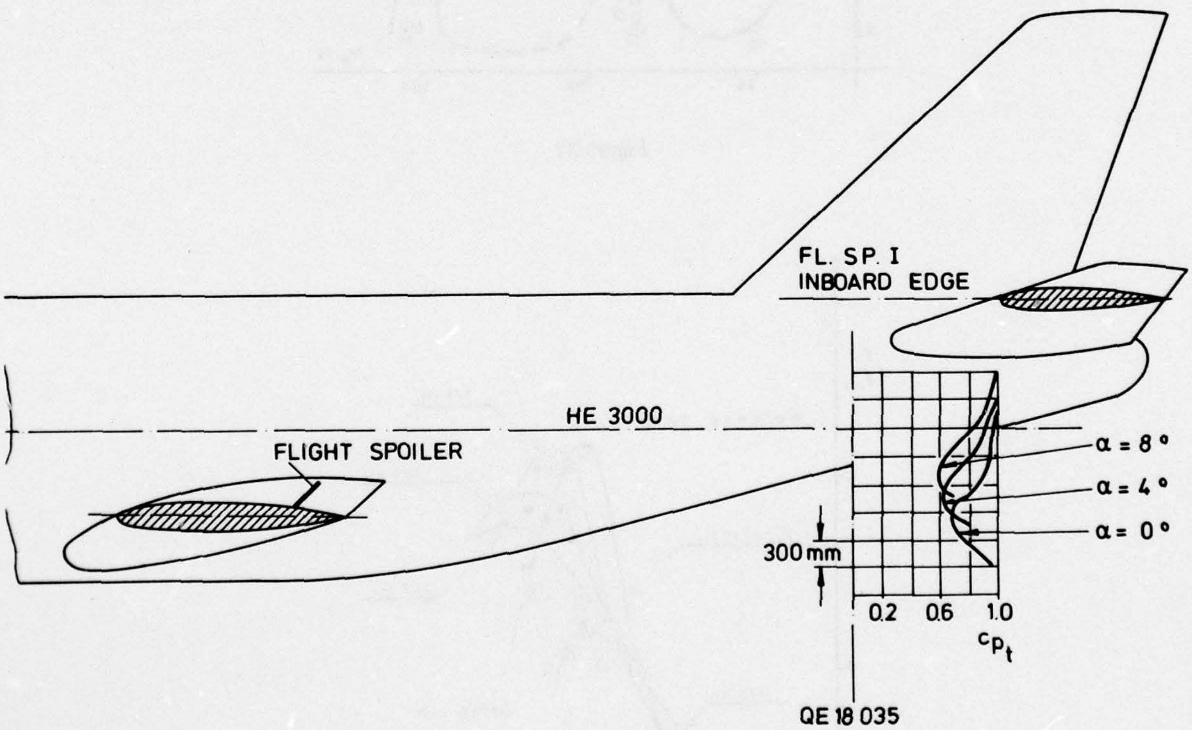


Fig.36 Flight spoiler wake at horizontal tail

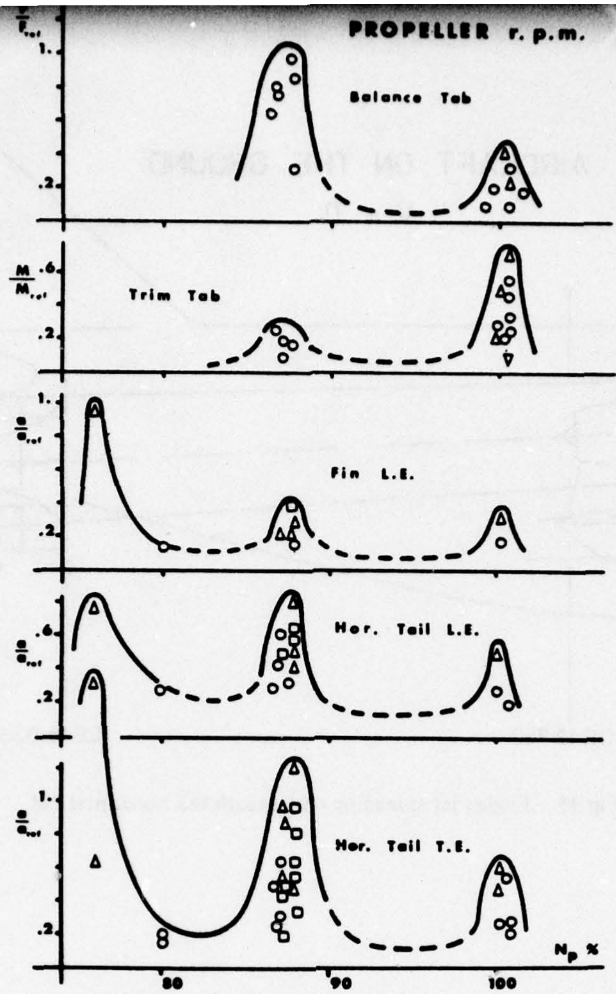


Figure 37

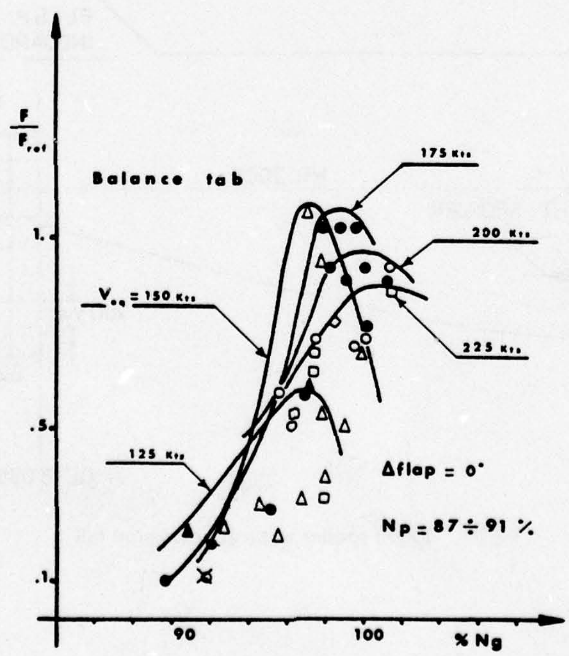


Fig.38 Influence of airspeed and turbine speed

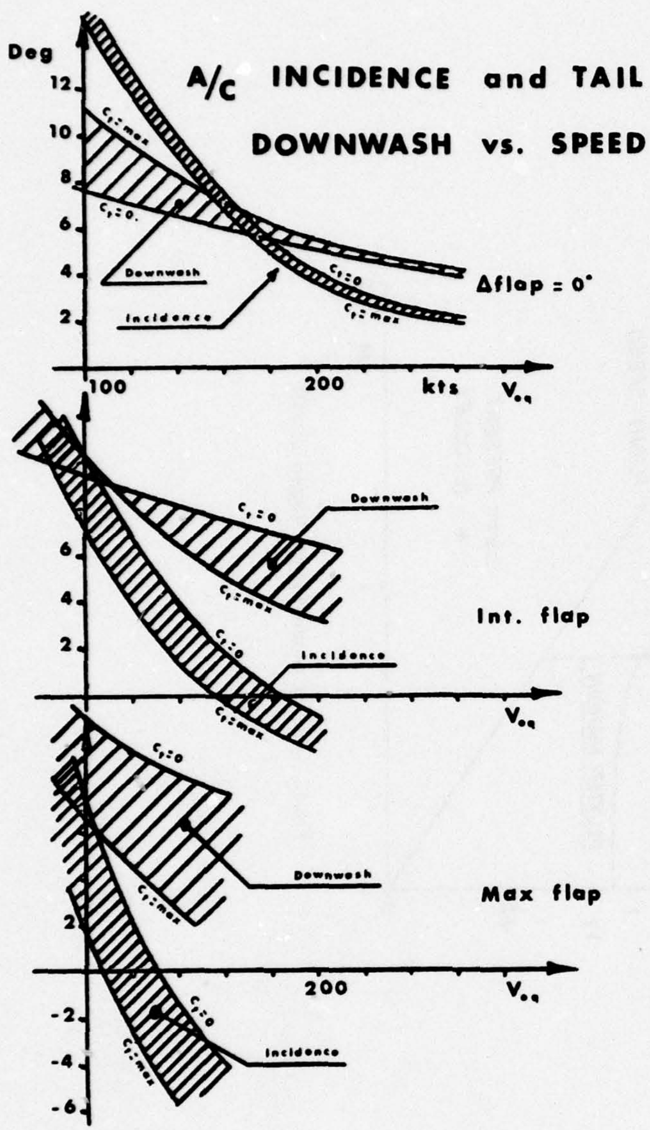


Figure 39

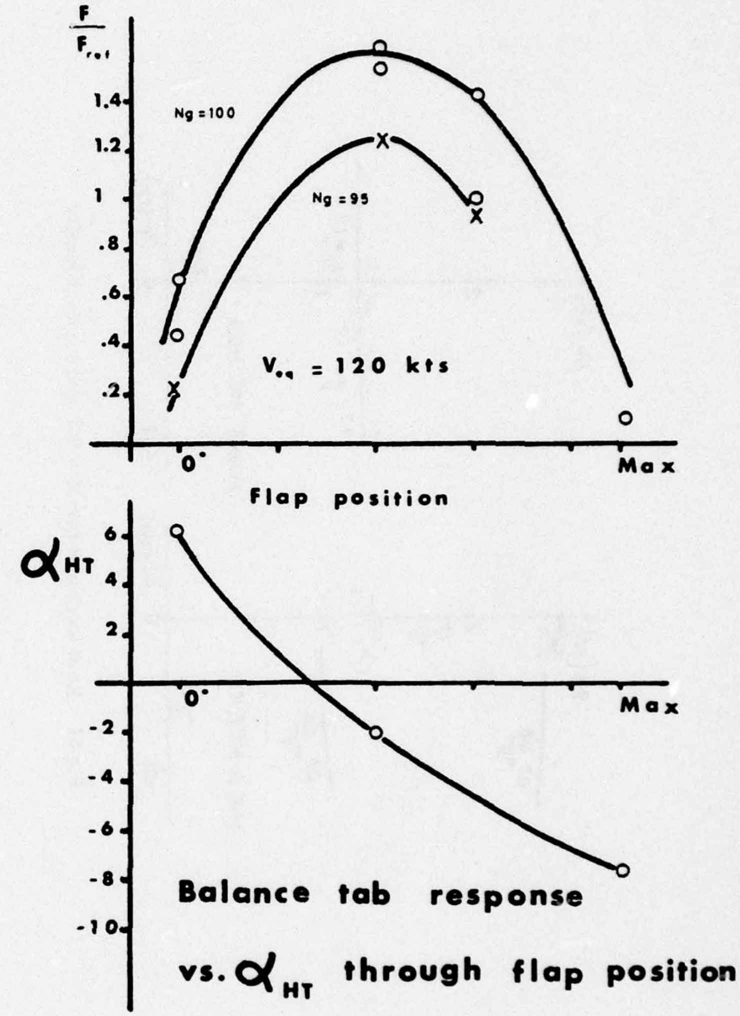


Figure 40

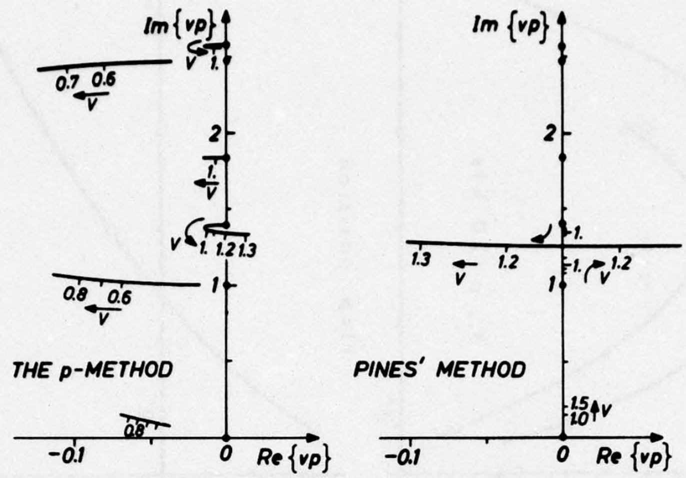


Fig.41 Root locus plot for M = 0.7 and sea level density

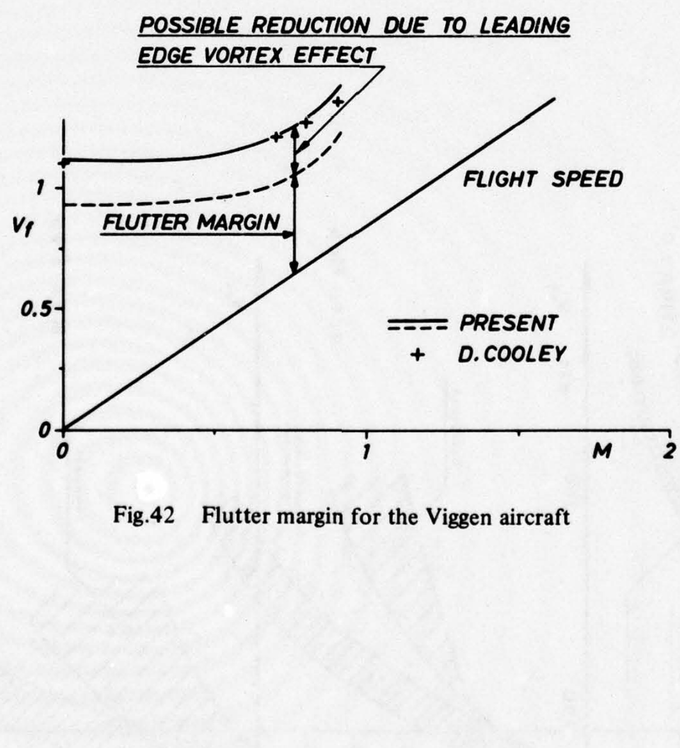


Fig.42 Flutter margin for the Viggen aircraft

SESSION II

PREFACE

The transonic speed range has been a critical aeroelastic design range for twenty years. Strong shocks or supercritical conditions increase unsteady pressure loadings, cause large phase shifts between wing motions and unsteady pressures, and result in low margins of safety for flutter, static aeroelastic and aeroservoelastic phenomena. No dependable analytical methods are yet available to predict unsteady airloads on three dimensional lifting surfaces or control surfaces. Heavy reliance has therefore been placed on flutter model tests and very approximate analyses, but wind tunnel tests produce unknown wall effects and do not match flight Reynolds numbers. Also, some model tests have revealed large reductions in flutter dynamic pressures in a narrow range of transonic Mach numbers. These reductions are not fully understood; therefore, results cannot be translated to full scale with confidence. Nevertheless, some remarkable progress is being made in the development of two dimensional non-viscous methods for predicting unsteady airloads up to $M = 0.9$. It was the purpose of this specialists' meeting to evaluate some of the latest methods, as well as some new and important measurements of airloads on an oscillating, thick supercritical airfoil. These papers and their references set the state-of-the-art and provide a groundwork for some judgments concerning the development of three dimensional engineering methods. They will help to establish priorities for the Structures and Materials Panel's Subcommittee on Aeroelasticity and Unsteady Aerodynamics, and will help to define standard configurations and parameters to be used in formal and informal joint experimental and analytical programs. The latter will be fully coordinated with the AGARD Fluid Dynamics Panel. Some thoughts on all of the above are outlined under Conclusions and Recommendations of this report.

W. J. MYKYTOW
Former Chairman
Subcommittee on Aeroelasticity
and Unsteady Aerodynamics

Technical Evaluation Report on Session II

TRANSONIC UNSTEADY AERODYNAMICS FOR AEROELASTIC PHENOMENA

by

W.J.Mykytow and J.J.Olsen
The United States Air Force Flight Dynamics Laboratory

SUMMARY

During the week of 18 April 1977, the AGARD Structures and Materials Panel conducted a specialists' meeting on "Unsteady Airloads in Separated and Transonic Flow." The specialists' meeting consisted of two sessions. Session I was titled "Airframe Response to Separated Flow." This report summarizes and evaluates the papers presented in Session II "Transonic Unsteady Aerodynamics for Aeroelastic Phenomena." Also included is an Appendix by Mr. H. C. Garner which contains remarks which did not appear in his paper, but have been the subject of considerable interest.

1. INTRODUCTION

Aeroelasticians historically have relied on linearized lifting surface theories to provide the aerodynamic forces necessary to analyze, design and optimize flexible aircraft structures for loads, deflections, stresses, stability, flutter and gust response. This has been due to the continuing lack of engineering methods which are useful for transonic speeds where, unfortunately, many aeroelastic problems are most severe. Aerodynamicists have recently made remarkable progress in predicting transonic steady and unsteady flows; however, the application of the newer transonic methods to aeroelastic problems has been quite limited. The purpose of Session II of the specialists' meeting was three-fold:

(a) To acquaint aeroelasticians in the international NATO community with the remarkable advances being made in transonic unsteady aerodynamics,

(b) To introduce aerodynamicists to the complex job of the aeroelastician with its requirement for enormous numbers of computations, where computational expense can be a severe problem,

(c) To foster future aeroelastic applications of the new transonic methods in order to improve the safety, efficiency and economy of aircraft structures.

With those purposes in mind, this Evaluation Report serves several functions:

(a) A condensation of the papers presented,

(b) An evaluation of the papers and meeting,

(c) A reference report to be utilized by both Fluid Dynamics Panel and Structures and Materials Panel.

(d) A framework for the coordination report by the Structures and Materials Panel to the Fluid Mechanics Panel at the latter's Fall 1977 meeting in Ottawa, Canada,

(e) A summary for flutter engineers not familiar with recent progress in unsteady aerodynamics.

2. TECHNICAL SUMMARIES

2.1 "Brief Overview of Transonic Flutter Problems," W. J. Mykytow

Brief comments were made on the history and chronology of flutter problems. Previous to 1957, they were of the control surface or tab type which were readily remedied by mass balance or local stiffness changes. In the brief interval of 1952-1956, problems occurred (Table I) involving transonic control surface buzz, wing-with-store flutter, T-tail flutter, all movable surface flutter, and fixed surface flutter. They remain critical flight safety problems today, and require considerable attention and cost to assure their avoidance. Lacking viable methods to predict transonic unsteady airloads, the reduced scale transonic flutter model has been the backbone for information in both research and aircraft developments. Research model tests (Figure 1) show the critical range extends up to free stream Mach numbers of 1.2 or higher. Dramatic drops in flutter speeds for swept back wings (Figure 2) occur at some Mach numbers, but analytical methods have not yet explained this behavior. Other figures in the report show model results used to avoid flutter and/or make design changes for fuselage-fin flutter, all movable surface flutter and wing-store flutter on a modern U.S. fighter. The British Aircraft Corporation provided the flight damping data shown on Figures 3 and 4 which illustrate the criticality of the transonic speed range from a flutter viewpoint. Thus, engineering flutter analysis methods have been urgently needed for twenty years for the entire range of configurations used on transonic aircraft. Methods developments should proceed jointly with many more unsteady pressure measurements. Correlations should be made with selected wind tunnel flutter model tests and limited flight test measurements. The Mach range should extend from 0.85 to 1.2, at least.

Dependable methods for predicting transonic unsteady airloads would reduce risks, increase confidence in early design optimizations and decisions, and reduce costs of model and flight tests.

2.2 "Unsteady Airloads on an Oscillating Supercritical Airfoil," H. Tijdeman, P. Schippers, and A.J. Persoon

Transonic unsteady pressure measurements are rare, but the limited number of such tests has contributed significantly to phenomenological understanding and prediction methods development. The authors present unsteady pressure measurements for a blunt nose, 16.5% thick supercritical airfoil oscillating in pitch for three flow conditions: fully subsonic flow, transonic flow with a well developed shock wave, and "shock-free" flow.

For subsonic flow, thin airfoil theory is in reasonable agreement with measurements for quasi-steady and unsteady pressures (Figure 5). Some differences are noticed at the nose because of thickness effects and at the rear from boundary layer effects.

Steady pressure measurements for transonic flow at higher lift coefficients show the strong shock. Noticeable shifts in its chordwise position occur for small angular changes. These shifts produce large quasi-steady pressure peaks not predicted by thin airfoil theory. Similar effects occur for unsteady pressures (Figure 6). The fairly high and sharp pressure peak shifts from in-phase to 90° out-of-phase due to lag in shock wave motion as frequency increases. The width and height of the pressure peaks reduce with increasing frequency (Figure 7). The phase jumps to 180° downstream of the shock.

The "shock-free" condition produces a wide chordwise region of supersonic flow over the upper surface, and a wide bulge in the pressure from 30-60% chord for quasi-steady and unsteady cases (Figure 8). Thin airfoil theory is completely inadequate. Much smaller changes are noticed for the slightly supercritical lower surface. The pressure in the supersonic bulge region decreases with increasing frequency, but a smaller and thinner peak near 65% chord due to a weak shock grows stronger as shock strength increases with frequency.

Lift and moment coefficients are obtained by integrating the measured unsteady pressures. They are compared to thin airfoil theory, which is again shown to be inapplicable for the strong shock case and shock free case (Figure 9) because of pressure peaks from shocks, pressure bulges, shock oscillations, shock lags, etc. The authors explain the behavior using qualitative arguments. The amplitude of the shock motion decreases with increasing frequency while the shock lag increases linearly because of a constant time lag associated with Kutta wave travel.

The authors evaluate the pulse behavior in pressure caused by oscillating strong shocks using a Fourier series expression for the observation point pressure and assuming that shock motion is sinusoidal and proportional to airfoil amplitude. Integration of pressures for lift and moment shows only a first harmonic in frequency for lift so that tube measurements using the first Fourier component are correct. Lift, therefore, behaves sinusoidally, but the moment has a shock amplitude squared term at a second harmonic of the frequency which would indicate irregularities in moment.

The authors point to the progress in solving nonlinear equations for unsteady transonic flow, but emphasize it has been only for inviscid flow. They discuss quasi-steady results, including the displacement effect of the boundary layer, and use an incidence corrected for wall interference. For the fully subsonic case, deviations from thin airfoil theory are due to thickness and incidence effects on the front of the airfoil and viscosity on the rear.

Viscosity has a large effect on steady flow and quasi-steady pressures for the transonic flow case with shock (Figure 10). When boundary layer and thickness effects are included, the more forward pressure peak on the upper surface is better predicted. Quasi-steady integrated coefficients show similar improvements and show that large increases (50%) in normal force coefficients were produced by thickness and incidence, but boundary-layer effects decrease the change by about the same order (-35%). Wall effects are noticeable. Calculations for a higher Reynolds number (30×10^6) do not show sensitivity compared to the low Reynolds number (2×10^6) used in the examples. For the shock-free condition, the quasi-steady theoretical design pressures for the upper surface show (Figure 11) a bulgy pressure distribution over the supersonic region, and agree reasonably well with experimental design condition data. The lower surface data also show an improvement relative to thin airfoil theory.

The authors fulfill a need for transonic unsteady pressure measurements, particularly for thick supercritical airfoils. Existing methods are evaluated and limitations of thin airfoil theory are delineated. Understanding of physical concepts is improved by the test data and comparisons with calculated data. Some of their important conclusions are:

- Inclusion of thickness, incidence and shock motions improve predictions.
- Discrepancies will remain from boundary layer effects which influence location of shocks and, therefore, overall unsteady airloads.

- Unsteady boundary layer research is in initial stages.
- Sophisticated methods will not be available in the near future.
- Engineering methods need to be employed for the interim.
- Small perturbation methods are attractive and should be evaluated.
- Wall effects on unsteady pressure measurements require assessment.
- Tests are required at higher Reynolds numbers.

2.3 "The Transonic Oscillating Flap; A Comparison of Calculations with Experiments," H. Yoshihara and R. Magnus

Research on unsteady boundary layer and separation effects is in primitive stages. Methods developments to predict transonic unsteady airloads are therefore limited to inviscid conditions. However, viscous effects are known to be important since they locate the shock position. The authors used displacement ramps to approximate viscous effects in two-dimensional, finite difference calculations based on the exact inviscid Euler equations. Calculations for the shock motions were made for the oscillating flap experiments reported by Tijdeman and Schippers. The mean position of the experimental shock was significantly further upstream, and the measured shock motions were significantly larger than calculated (Figure 12). Also, the shock briefly disappeared for a brief interval when the shock velocity caused subcritical flow upstream of the shock. In view of the above, a mis-match of pre-shock pressures (Figure 13), previous success with the inviscid procedure, and an excellent agreement between steady calculations at M = 0.854 and experimental results at M = 0.975 for zero flap angle (Figure 14), it was suggested that the cause is in the experimental simulation due to inadequate test section length or wall interference. Mach number 0.875 test data were then assumed to be applicable to M = 0.854. Inviscid calculations showed a noticeably improved agreement (Figure 15) in shock location and displacement with sinusoidal flap position. The authors state that, while pre-shock pressure mis-match seriously influenced shock behavior, the effect was far less serious on unsteady lift and moments used for flutter applications. They also suggest tests in larger wind tunnels and the use of airfoil embedded dynamic pressure transducers to obtain higher harmonics of the pressure signals.

The approximation of viscous effects by use of a viscous ramp has produced rewarding qualitative improvements in location and amplitude of oscillatory shock motions for the case considered. Interpretations of wind tunnel data versus free flight conditions has been questioned and improved.

2.4 "Numerical Calculation of Unsteady Transonic Flows," A. Lerat and J. Sides

There is a continuing need to reassess the assumptions made in earlier computations. The hopefully small errors involved in satisfying the tangency conditions on a coordinate axis or on the mean steady position of an airfoil may turn out to be too large to accept. Further, the often-used isentropic flow assumptions may come into question for many aeroelastic problems. These authors are developing computational methods to address these types of questions.

They solve the full hyperbolic Euler equations for unsteady transonic flow, satisfying the tangency conditions at the exact instantaneous surface positions, the Kutta conditions at the trailing edge, and allowing nonisentropic flow. Initial conditions are taken from the steady flow over the airfoil in its initial position. At infinity, the initial steady flow is retained at great distances, except that in the far-wake, the derivatives of velocity and density with respect to distance are set equal to zero. They map the exact space domain onto a fixed simple domain by a product of two transformations; the first specifies the rigid body motion of the airfoil, and the second maps the exterior part of the flow onto a rectangle. After transformation, the system of equations is recast in conservation form and solved by several variants of MacCormack's scheme, depending on the flow region under consideration. The solution method is second order accurate in the transformed spatial variables, and the time step is restricted by a Courant-Friedrichs-Lewy-like stability condition. They use an artificial viscosity and mesh refinement to improve the capture of shocks.

Results are presented for an NACA 0012 airfoil oscillating in pitch about an axis at the 0.25 chord, a reduced frequency of $\frac{c\omega}{V} = 10.0$ and $M_\infty = 0.8$. The outer boundary is held at about 6.0 chord lengths. The steady state pressures were compared to similar results of Garabedian and Korn, Figure 16, noting that the shock jump results of Sides and Lerat satisfy the Rankine-Hugoniot conditions with good accuracy. They also capture the slip line from the trailing edge to downstream infinity. Unsteady periodic results were obtained after two cycles of motion, revealing very little shock motion due to the high frequency, Figure 17. For a mesh of 144 x 20 points, oscillatory computations consumed 30 minutes per period of oscillation on the Univac 1110 computer.

2.5 "Efficient Solution of Unsteady Transonic Flows About Airfoils," W. F. Ballhaus and P. M. Goorjian

Explicit finite difference schemes may be inefficient. The authors apply efficient, conservative-form, alternating-direction, implicit finite difference procedures to the low frequency, transonic small disturbance approximation of Eulers equations. The time step is based on accuracy rather than stability considerations.

Results are obtained for the oscillating flap experimental case reported by the Netherlands National Aerospace Laboratory. The three types of shock wave motions (upstream-propagating shock motions, interrupted shock wave motions, and sinusoidal shock wave motions observed as high subsonic free stream Mach numbers are increased) are simulated by the numerical results. Unexpectedly good agreement is obtained with the more exact and much longer calculations of Magnus and Yoshihara. The authors also compute effects of a solid wall ($\phi_y = 0$) and a free jet boundary ($\phi_x = 0$). They show that the shock wave on the airfoil in the solid wall case is stronger (Figure 18) than the free air case, and the free jet shock wave is weaker still. Since such shock motions were reported for higher Mach numbers in the NLR tests than in the calculations, the wind tunnel wall effect could be one that weakens shock waves. Figure 19 for M = .865 shows an oscillating shock for free air conditions and an interrupted shock for free jet conditions.

Other applications are made to a pitching airfoil. The flow field equations and the structural equations are simultaneously integrated in time to demonstrate stable and unstable cases. Non-sinusoidal pitching moment behavior results from large amplitude motions causing large shock wave excursions.

The indicial response method using Duhamel's integral is applied to the three shock motion cases produced by NLR's oscillating flap experiment. Compared to the time integration method, the linear perturbation assumption produces valid lift and moment cyclic variations for weak shock waves corresponding to the upstream propagating waves and the interrupted shock wave motion cases at the two lower Mach numbers. However, noticeable differences in lift and moment calculations are apparent (Figure 20) between indicial and time integration methods for the oscillating strong shock case at the higher Mach number.

Implicit finite difference schemes are more efficient and less expensive than explicit schemes. Therefore, they hold considerable promise in further developments. Time integration techniques would cause increased costs and a major change to the aeroelastician's frequency domain analyses, but they could include nonlinear effects.

2.6 "Application of a Finite Difference Method to the Analysis of Transonic Flow Over Oscillating Airfoils and Wings," W.H. Weatherill, J.D. Sebastian, and F.E. Ehlers

A natural extension of the currently accepted linear unsteady aerodynamic methods is to calculate the nonlinear steady flow with shock waves, and then linearize the unsteady calculations about the nonlinear steady results. If successful, this technique can be adapted to conventional flutter analyses with a minimum of complication, but somewhat additional expense.

The authors separated the small disturbance, high frequency, unsteady transonic velocity potential into a steady term and an unsteady term (linearized about the nonlinear steady solution).

$$\phi = \phi_0(x,y) + \phi_1(x,y)e^{i\omega t}$$

The steady flow and unsteady response due to simple harmonic airfoil oscillations were calculated by relaxation techniques similar to the original concepts of Murman and Krupp.

Tangency conditions were satisfied on the line y = 0, and the Kutta condition was satisfied at the trailing edge. In specifying boundary conditions on the mesh boundaries, the upstream and downstream conditions were specified in terms of the unsteady pressure potential:

$$\phi_{1_x} + i\omega\phi_1$$

and the value of ϕ_1 was specified at one point on the upper boundary and one point on the lower boundary.

In obtaining relaxation solutions of the large system of difference equations, substantial numerical stability problems were encountered with purely subsonic as well as mixed subsonic-supersonic flows. The numerical instability depended on the mesh dimensions, growing more serious as frequency increased or Mach number approached 1.0.

To gain insight into the stability problems, the authors examined finite-difference solutions of Helmholtz' equation and considered various types of outgoing-wave

and porous wall conditions for the transonic problem. They found that no combination of the boundary conditions considered gave a converged relaxation solution above a certain low reduced frequency which would depend on the Mach number and mesh geometry. They also examined Carlson's stretched coordinates in combination with homogeneous and outgoing wave boundary conditions. They noted little beneficial effect on the numerical stability problems, but better agreement with linear theory using outgoing wave conditions. An examination of direct solutions of the large matrix equations revealed markedly improved numerical stability, but poor accuracy at frequencies above the frequency at which relaxation techniques became unstable, possibly due to the relatively crude mesh used.

They concluded that they had observed numerical instabilities in the relaxation solutions for unsteady flow with free stream Mach number greater than 1.0, as well as less than 1.0.

Using steady state results by Ballhaus and Bailey, the authors calculated the unsteady pressures on a rectangular wing, oscillating in pitch about the leading edge at $M_\infty = 0.875$, and at a reduced frequency of $\frac{c\omega}{2V} = 0.06$ (Figures 21,22). They expressed some doubt about the validity of their 3D results at the 57% span station, which seemingly had a stronger shock than the root station.

The linearized unsteady method, if successful, has the great advantage that it could be easily incorporated into existing aircraft flutter engineering practice which is normally performed by seeking eigensolutions of linear aerodynamic and structural equations in the frequency domain. Numerical instability difficulties in the relaxation solutions need to be resolved, however, by an efficient procedure.

2.7 "Numerical Solution of the Unsteady Transonic Small Disturbance Equations," M.M. Hafez, M.H. Rizk, and E.M. Murman

This paper continued the examination of the numerical instability problems encountered recently in relaxation solutions of the transonic unsteady equations. Of additional interest was their development of consistent shock jump conditions which allow shock movement in the first order unsteady solutions.

The low frequency unsteady equations for ϕ_1 reduce to a form

$$(K-\phi_{0x})\phi_{1xx} + \phi_{1yy} - \phi_{1x}\phi_{0xx} = i\omega\alpha\phi_{1x}$$

$$\phi_{1y}(x,o) = h_1^1(x)$$

Using type-dependent finite-difference schemes, they obtained the system of algebraic equations

$$[A(\phi_0)]\{\phi_1\} = \{f\}$$

where [A] is the system matrix, $\{\phi_1\}$ is a vector of the unknown potentials, and {f} is a vector for the nonhomogeneous terms and boundary conditions. The authors note relaxation procedures will only converge if the matrix [A] is positive-definite. They proposed premultiply [A] by its conjugate transpose to obtain

$$[A]^*[A]\{\phi_1\} = [A^*]\{f\}$$

where [A*][A] will be positive-definite at non-singular points, but unfortunately will have a bandwidth of almost twice that of [A] and can even be more ill-conditioned than [A]. Relaxation procedures will then usually converge using [A*][A], even if they converge slowly.

The authors demonstrated their contention by solving the one-dimensional problem

$$\phi_{xx} + \omega^2\phi = 0 \qquad 0 < x < 1$$

$$\phi(0) = 1/2 \qquad \phi(1) = 1$$

exactly, by relaxation of $[A]\{\phi\} = \{f\}$ and of $[A]^*[A]\{\phi\} = [A^*]\{f\}$, and by direct matrix solution of both systems (Table 2).

In developing consistent shock-jump conditions for first-order treatment of unsteady flows, the authors considered the potential expansion:

$$\phi = \phi_0(x,y) + \epsilon\phi_1(x,y)e^{i\omega t} \ldots$$

and the consistent shock motion

$$V_s(y_1 t) = X_0(y) + \varepsilon X_1(y) e^{i\omega t} \ldots$$

They arrive at the following conditions

$$\text{Jump} \quad [\![\phi_0]\!] = 0$$

$$\text{Average} \quad \langle K - \phi_{0_x} \rangle = -\left(\frac{dX_0}{dy}\right)^2 \quad \text{at} \quad x = X_0(y),$$

$$[\![\phi_1]\!] = -X_1 [\![\phi_{0_x}]\!],$$

and the equation for the unsteady shock movement, X_1,

$$i\omega\alpha X_1 = \langle \phi_{1_x} + X_1 \phi_{0_{xx}} \rangle - 2\frac{dX_0}{dy}\frac{dX_1}{dy} \quad \text{at} \quad x = X_0(y)$$

They gave simplified forms for a normal shock, developed a shock-fitting procedure for the ϕ_1 problem, computed a one-dimensional example, and credited an alternative approach by Nixon which used the method of strained coordinates.

While the solution of the numerical stability problems in the iterative processes is seen to require further work, the incorporation of small shock motion conditions into the first order unsteady results is shown to be a conceptually simple process which can be performed after the ϕ_1 solution is available.

2.8 "A Practical Framework for the Evaluation of Oscillatory Aerodynamic Loading on Wings in Supercritical Flow," H. C. Garner

The author describes an economic and approximate engineering method based on nonlinear steady pressures (taken from small perturbation theory or from static pressure measurements) and theoretical linear oscillatory loadings. The three dimensional method's running time is small compared to that required to obtain results for the thin-plate wing from linear theory at uniform (steady) incidence and in oscillation, or steady state data from a finite-difference transonic small perturbation method. The approximate method could provide indications of main influences economically for free stream Mach numbers below 0.9. The approach is based on three assumptions: (a) the use of a one-dimensional form of Bernoulli's equations resulting in a one-dimensional expression for oscillatory pressure coefficient and ignoring influence of the lateral component of mean flow; (b) the assumption that the ratio of the local oscillatory chordwise component of velocity to its value in the quasi-steady ($\omega = 0$) case is the same as the corresponding ratio from linear theory; and (c) the assumption that the ratio of the quasi-steady rate of change of surface pressure to its linearized theoretical rate of change is the same for all modes of deformation or displacement. Applications to a rigid half wing oscillating about a swept axis reproduce qualitatively the large and rapid changes in amplitude and phase of the local loadings measured in the vicinity of the shock wave (Figures 23, 24). Linear theory predicts a linear phase variation indicating a phase lag on the forward chord and a lead on the aft chord. The approximate method shows greater lags due to the delay in upstream propagation, and also higher lead angles on the aft chord, but both values are lower than measured. The increased phase lag needs to be better simulated and is more important, as pointed out by the author, since it occurs in regions of higher loadings. The author later questions the inadequacy of the second assumption relative to time delay even though attenuation within the mean flow has been taken into account. Generalized forces for a pitching and heaving wing are presented for linear theory, the approximate method using static data from transonic small perturbation theory, and the approximate method using measured static data (Figure 25). The author states, "On the hypotheses that at full scale, the steady-state results would lie between the predictions of the linearized and transonic small perturbation theories and that the semi-empirical method is representative of the wind tunnel, it could be deduced that experimentally determined (tunnel) aerodynamicsforces would differ from their full scale values by as much as 50%."

Other applications show spanwise loadings for the wing oscillating about the swept axis. Results using transonic small perturbation theory give higher in-phase loadings than linear theory, the method using measured steady data, and experiment, respectively (Figure 26). The latter gives lowest values inboard of 80% semi-span. The measured quadrature or imaginary loading versus span (Figure 26) is more negative than the three predicted values. This is due to the larger phase lags at forward chord regions. Predicted oscillatory centers of pressure versus span (Figure 27) are more aft for the approach employing transonic small perturbation static data. Those obtained

from linear and semi-impirical (measured static pressure) methods are less aft. Experimental measurements show the most forward positions. Aft aerodynamic centers raise flutter speeds so that predictions based on inviscid supercritical flow could be over-optimistic, but those based on tests at low Reynolds number may be too conservative.

The author mentions an attempt to extend the method to oscillating control surfaces with little success due to the inadequacy of the second assumption. He also foresees the incorporation of the three dimensional boundary layer growth into Albone's finite difference technique. This would, as he states, lift the restriction to wind tunnel Reynolds numbers on the one hand and to infinite Reynolds numbers on the other, so that full scale unsteady aerodynamic loads would be approximated.

In addition to some of the comments made above, the author also suggests for a future framework: assessment of approximations; more rigorous theoretical attacks on the problem of inviscid unsteady three dimensional flow; basic research on unsteady boundary layer and shock wave interaction; and detailed wind tunnel investigations of oscillatory surface pressures on rigid and flexible models.

This approximate method appears to give fair agreement with some experimental data. Additional applications are needed to assess its utility and limitations. Further developments and evaluations of approximate engineering methods will likely be required before validated and accurate three dimensional methods are established for practical engineering applications.

3. CONCLUSIONS AND RECOMMENDATIONS:

With the minimum margins of safety and lack of a reliable method to predict unsteady airloads, the transonic speed region remains the most critical and dangerous speed region for aeroelastic aircraft design. Many of the following conclusions and recommendations reflect the views offered in References 1-8. Others represent opinions of the present authors as they occurred in reading and listening to the papers or from other experience. Some interesting comments and a Venn diagram by H. C. Garner on the state-of-the-art are reproduced in the Appendix.

3.1 Basic Physical Principles

The steady flow field has very considerable effects on time-dependent phenomena at transonic speeds, and they must be considered. Large changes in the amplitude and phase of the unsteady pressures occur for airfoils with supercritical regions and strong shocks. Linear thin-plate theories are completely inadequate under such conditions. Therefore, thickness, camber, incidence, oscillatory amplitudes and shock motions must be considered.

Viscous effects and wind tunnel wall effects assume increasing importance at transonic speeds, and must be evaluated. Boundary layer effects on unsteady pressures can be on the same order and opposite to thickness effects.

Unsteady boundary conditions posed by porous walls remain a fertile and an essential area for theoretical, numerical and experimental investigations and explanations.

Quasi-steady results in the sense defined by Tijdeman can give insight into the physical mechanisms of low frequency transonic unsteady flows.

The transonic unsteady aerodynamic behavior of wings with supercritical airfoils has the potential to pose serious surprises for the aeroelastician if he relies on thin plate linear theories.

3.2 Aerodynamic Computational Methods

Recent emphasis has been on transonic methods for free stream flows up to $M_\infty = 0.9$. Methods must be extended to $M_\infty = 1.2$ or higher since aeroelastic safety remains low up through the low supersonic speed region.

Explicit solution of the large disturbance Euler equations remains a very expensive technique, even for two dimensional airfoils. The solutions may be considered as "exact" inviscid solutions, but extension to three dimensional wings remains beyond the near future. The incorporation of approximate viscous effects through "viscous ramps" shows some promising qualitative improvement, but requires additional exploration.

Satisfaction of the large disturbance tangency conditions on the instantaneous airfoil surface has been shown to be not appreciably more difficult than use of the mean airfoil position. Dr. R. Magnus has shown (in an investigation to be reported) relatively large effects on phase changes.

Additional emphasis on transformation techniques which cluster solution points in regions of large gradients will be most beneficial in reducing computing costs.

Severe numerical stability problems have limited the usefulness of the

relaxation methods in combination with frequency-expansions. Several fixes have been tried with various degrees of success. Solution of these problems would be most beneficial since the frequency-expansion methods can be incorporated in conventional flutter analyses with minimum procedural difficulty, but considerable expense.

The correct unsteady shock-jump conditions should be included in the frequency-expansion techniques to account for shock motions and their large effects on unsteady airloads.

Implicit techniques to solve the low frequency (and high frequency) transonic unsteady equation can be much less expensive than explicit techniques. Shock motions occur naturally; however, time-integration methods will require that aeroelastic analyses be done in the time domain, a substantial alteration to the current methods which usually are done in the frequency domain.

Engineering approaches are being developed which use steady state numerical or experimental results as a base in conjunction with flat plate lifting surface theories. These hold some promise of lessening the gap between the aerodynamicist's computational methods and the aeroelastician's appetite for response calculations. Additional research is necessary to extend the methods beyond $M_\infty = 0.9$ and to apply to wings with control surfaces.

3.3 Experimental Data

Provision of experimental data is a major priority.

There is a general lack of experimental data on all transonic unsteady aerodynamic effects, particularly on viscous and wind tunnel wall effects. Transonic unsteady pressure measurements on oscillating models in high Reynolds number facilities are urgently needed.

Also, there is a dearth of experimental unsteady pressure data for three dimensional configurations in the transonic range. While some tests are in process, information is minimal and severely limits methods development, evaluations, and improvement. Quasi-steady data, however, reveal important information.

Excellent research and aircraft development applications of transonic unsteady pressure measurements have been made, particularly in Europe, on a very limited number of configurations. The U.S. continues to lag behind the state-of-the-art in this area.

Two dimensional flutter model tests could provide significant basic information on many unexplained flutter results.

Some excellent transonic flutter data on well defined models exist. These wind tunnel data should be compared with predictions from emerging methods and vice versa.

Limited flight measurements of unsteady pressures could be valuable for evaluating Reynolds number and wall effects.

The "in situ" transducer and "tube-type" measurement systems seem to give essentially identical results for the first harmonics of unsteady pressures on oscillating models. Further development of the measurement systems for large-scale unsteady applications (up to 500 measurements on each of several different models) requires extensive studies of tradeoffs between accuracy and costs.

3.4 Aeroelastic Applications

The steady transonic computer codes recently developed are being used extensively. However, applications of unsteady transonic load computer codes to aeroelastic investigations are lagging due largely to the lack of three dimensional methods. Such extensions are necessary. The inclusion of viscous effects will be difficult.

The AGARD Structures and Materials Panel should encourage aeroelastic applications of existing and emerging transonic airload prediction methods and give high priority to exchanges of information in this area.

Engineering or approximate methods will find considerable interim use. Several are under development and require applications to flutter problem cases.

The SMP might wish to review developments and applications at a meeting in about two years.

In the meanwhile, lacking methods for realistic predictions of transonic unsteady airloads, <u>no</u> realistic transonic flutter analyses and <u>no</u> realistic transonic strength/stiffness design optimization studies can be made with confidence. Heavy reliance on more costly aeroelastic models will continue.

It is conceivable that transonic unsteady wall interference effects could cause an aeroelastic wind tunnel model to yield <u>unconservative</u> predictions of flight safety.

3.5 Cooperation

National and international cooperation and coordination has resulted in informal joint programs and accelerated progress. Standard configurations and parameters should be defined to provide comparisons on a limited number of standard examples. Such action is now being implemented by the Aeroelasticity and Unsteady Aerodynamics Subcommittee of AGARD's Structures and Materials Panel, and is strongly indorsed. Very close coordination with the Fluid Dynamics Panel is essential in view of the highly coupled nature between steady and unsteady transonic flow and the accelerated progress, particularly for the steady flow.

4. REFERENCES

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[3] Yoshihara, H. and Magnus, R. "The Transonic Oscillating Flap; A Comparison of Calculations with Experiments"

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[6] Weatherill, W.J.; Sebastian, J.D.; and Ehlers, F.E. "Application of A Finite Difference Method to the Analysis of Transonic Flow Over Oscillating Airfoils and Wings"

[7] Hafez, M.N.; Rizk, M.H.; and Murman, E.M. "Numerical Solution of the Unsteady Transonic Small-Disturbance Equations"

[8] Garner, H.C. "A Practical Framework for the Evaluation of Oscillatory Aerodynamic Loading on Wings in Supercritical Flow"

Table 1. Some Past Flutter Incidents.

	1947 - 1951	1952 - 1956
TABS: CONTROL SURFACES	11	8
WING WITH EXTERNAL STORES	1	6
AUTOPILOT COUPLING	1	
TRANSONIC BUZZ RELATED		21
T-TAIL		1
ALL MOVABLE SURFACE		4
FIXED SURFACE - BENDING - TORSION		1

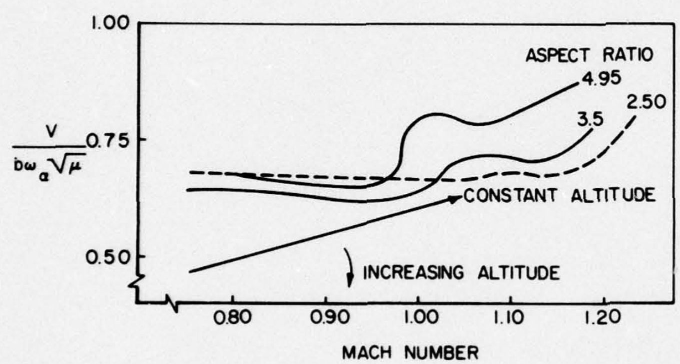


Fig.1. Transonic Flutter Model Tests for Unswept Cantilever Wings.

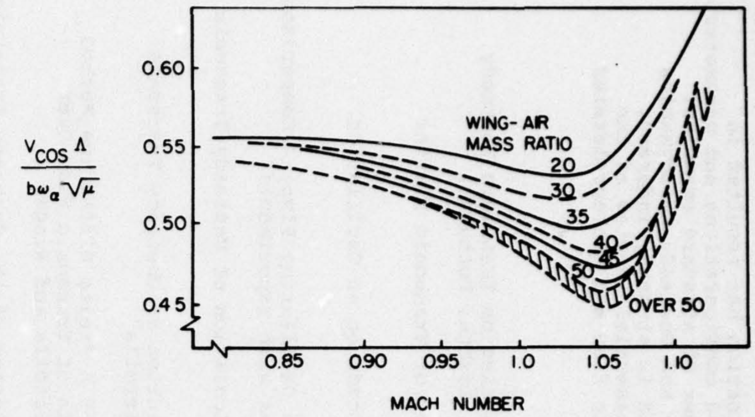


Fig. 2. Transonic Flutter Model Tests on Swept Cantilever Wings.

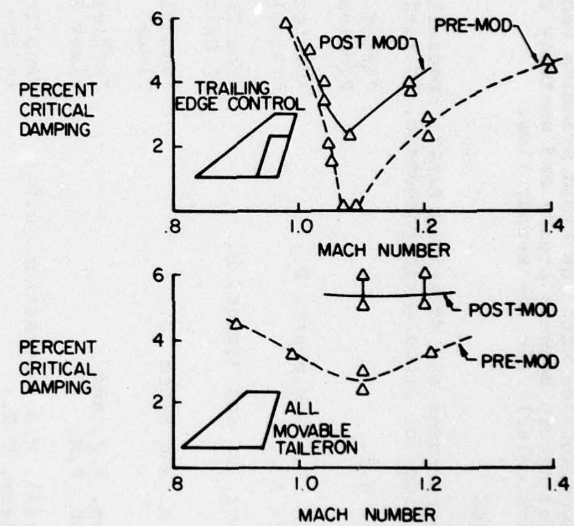


Fig. 3. Transonic Flutter Damping.

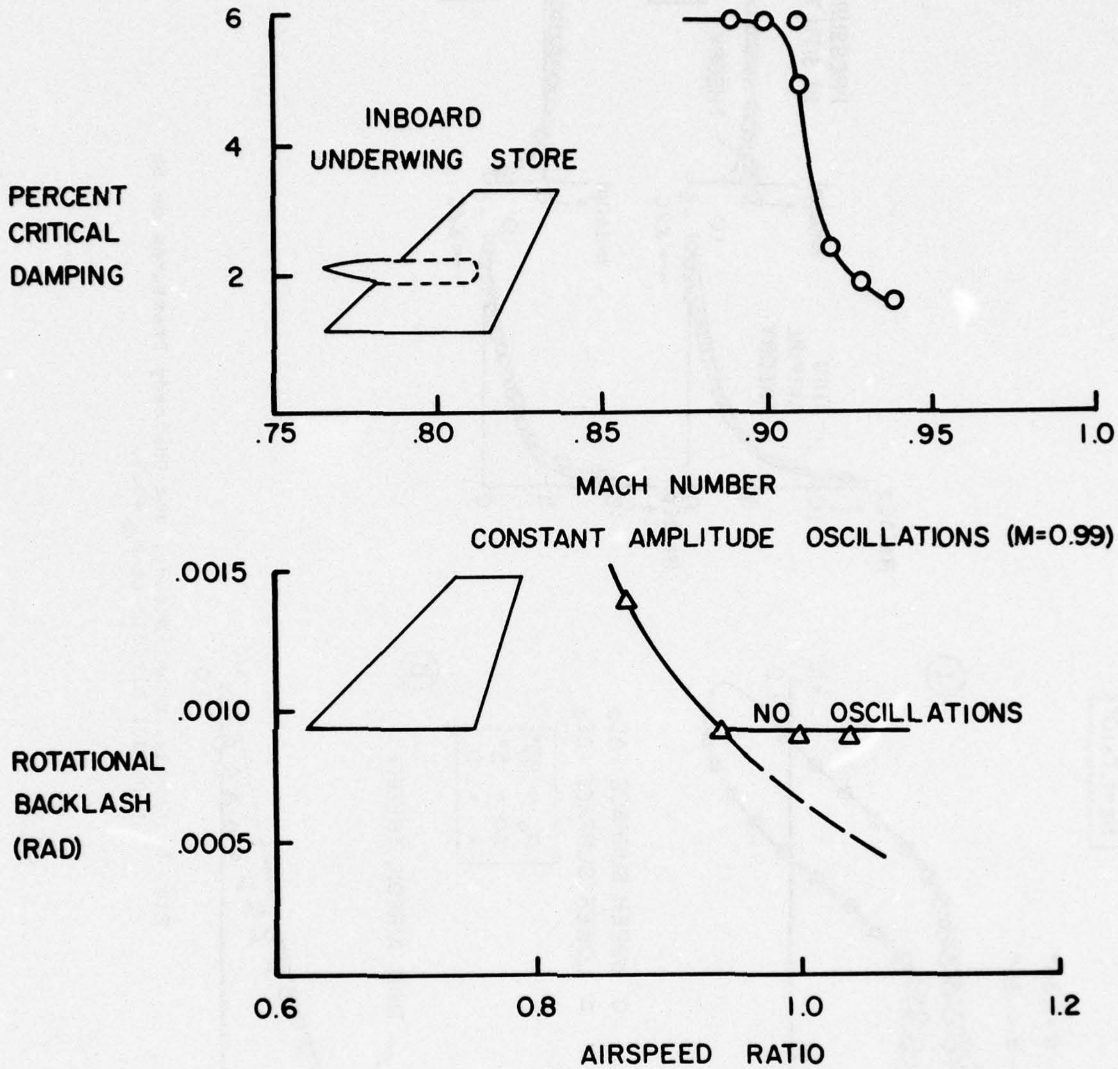


Fig. 4 Transonic Flutter Damping and All-Movable Tail Rotational Backlash.

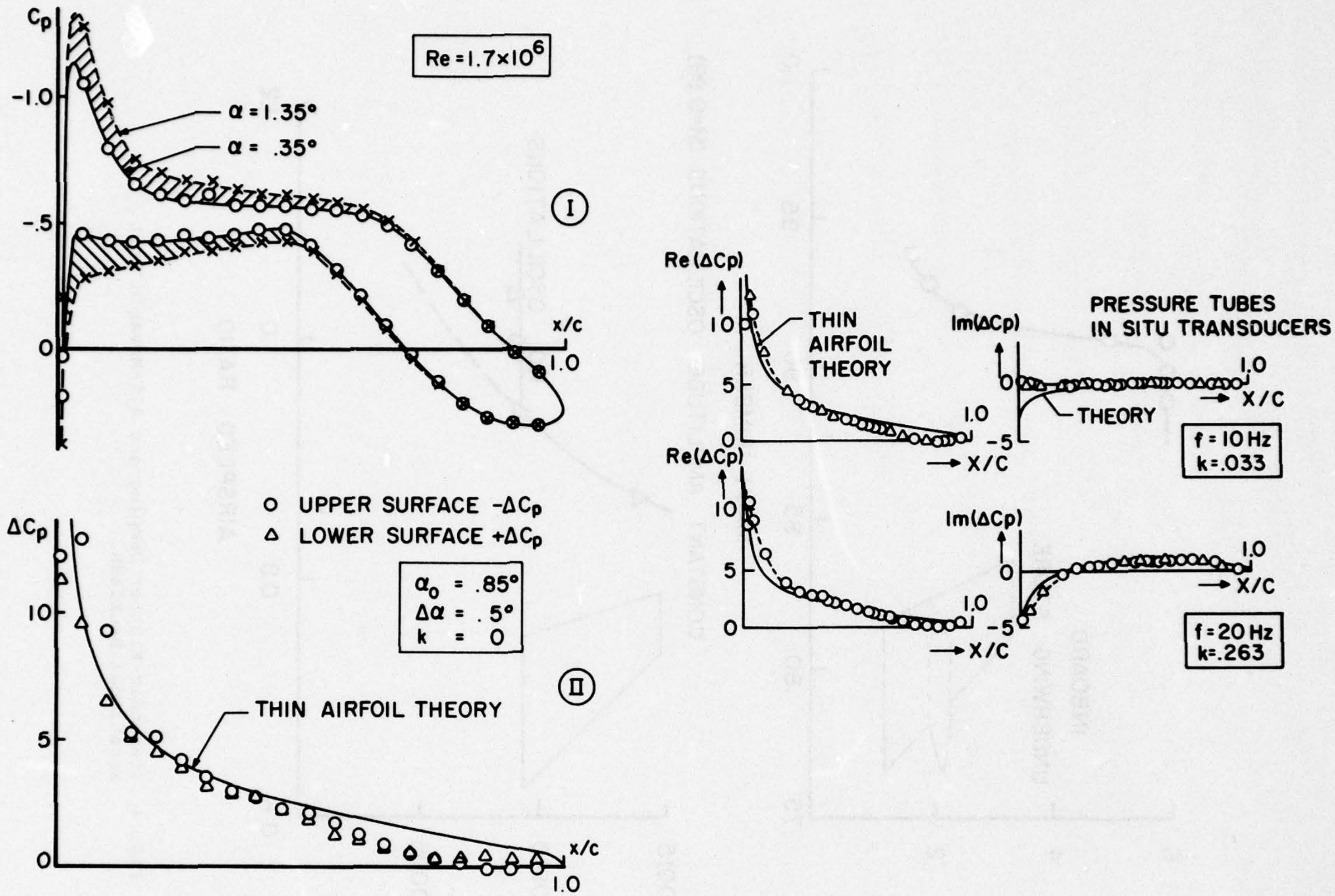


Fig. 5 Steady, Quasi-Steady, and Unsteady Pressures on an NLR 7301 Airfoil at $M_\infty = 0.5$.

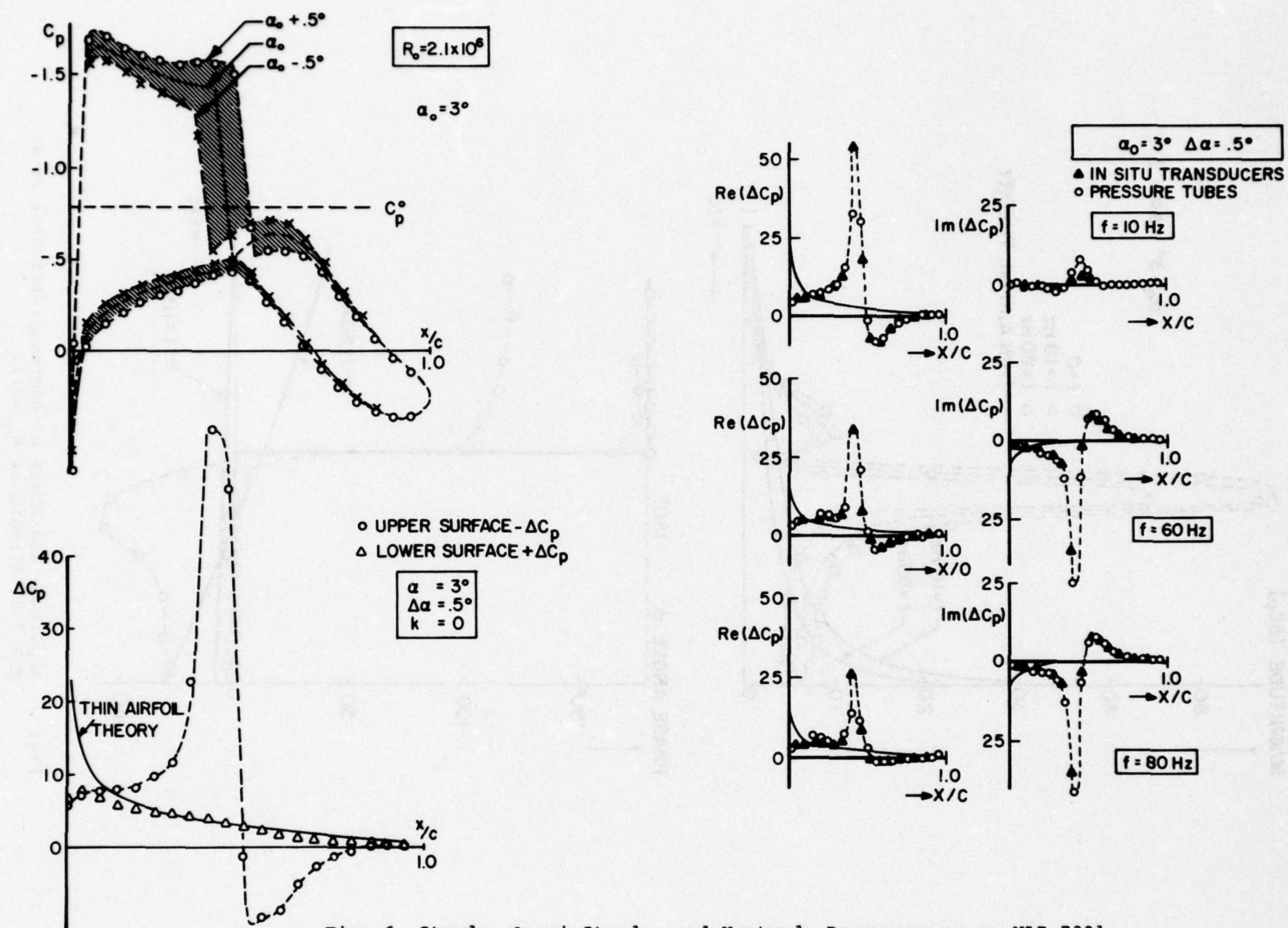


Fig. 6 Steady, Quasi-Steady, and Unsteady Pressures on an NLR 7301 Airfoil at M_∞=0.7.

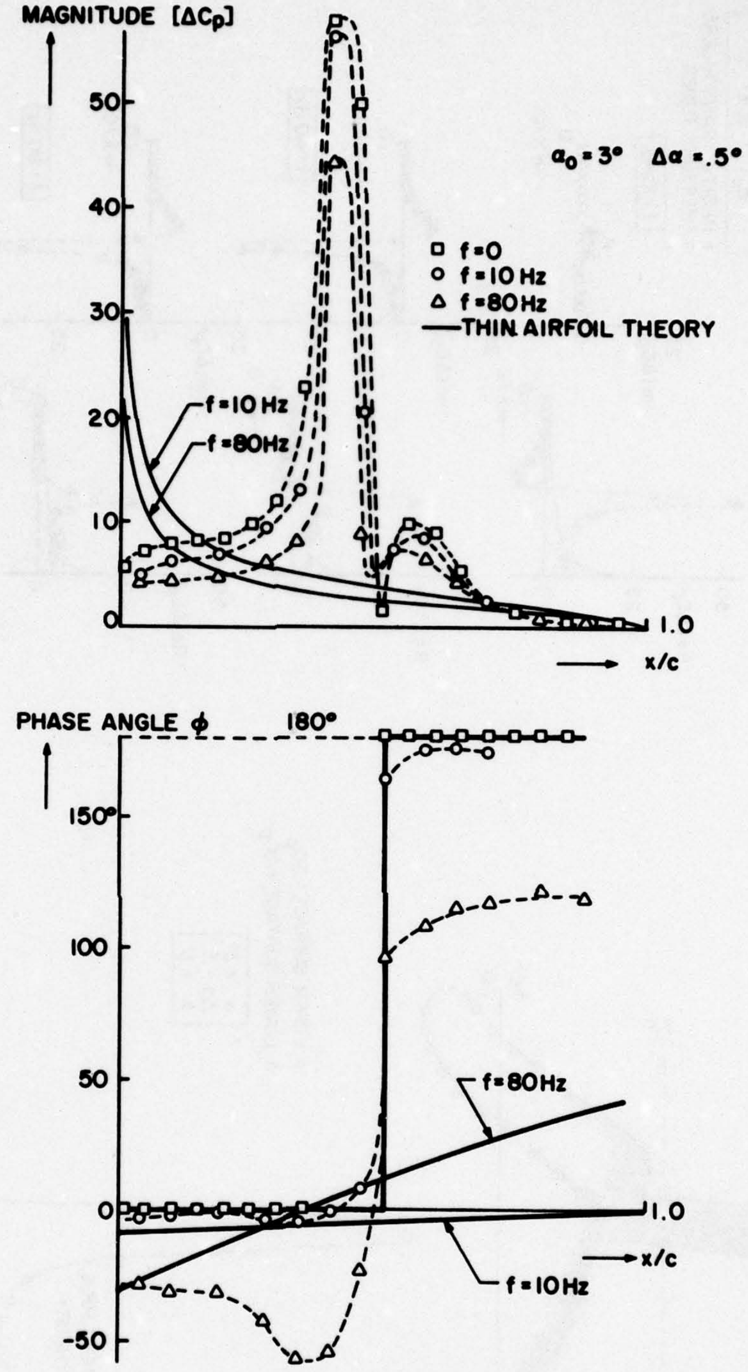


Fig. 7 Amplitude and Phase of Unsteady Pressures on an NLR 7301 Airfoil at M_{∞} =0.7.

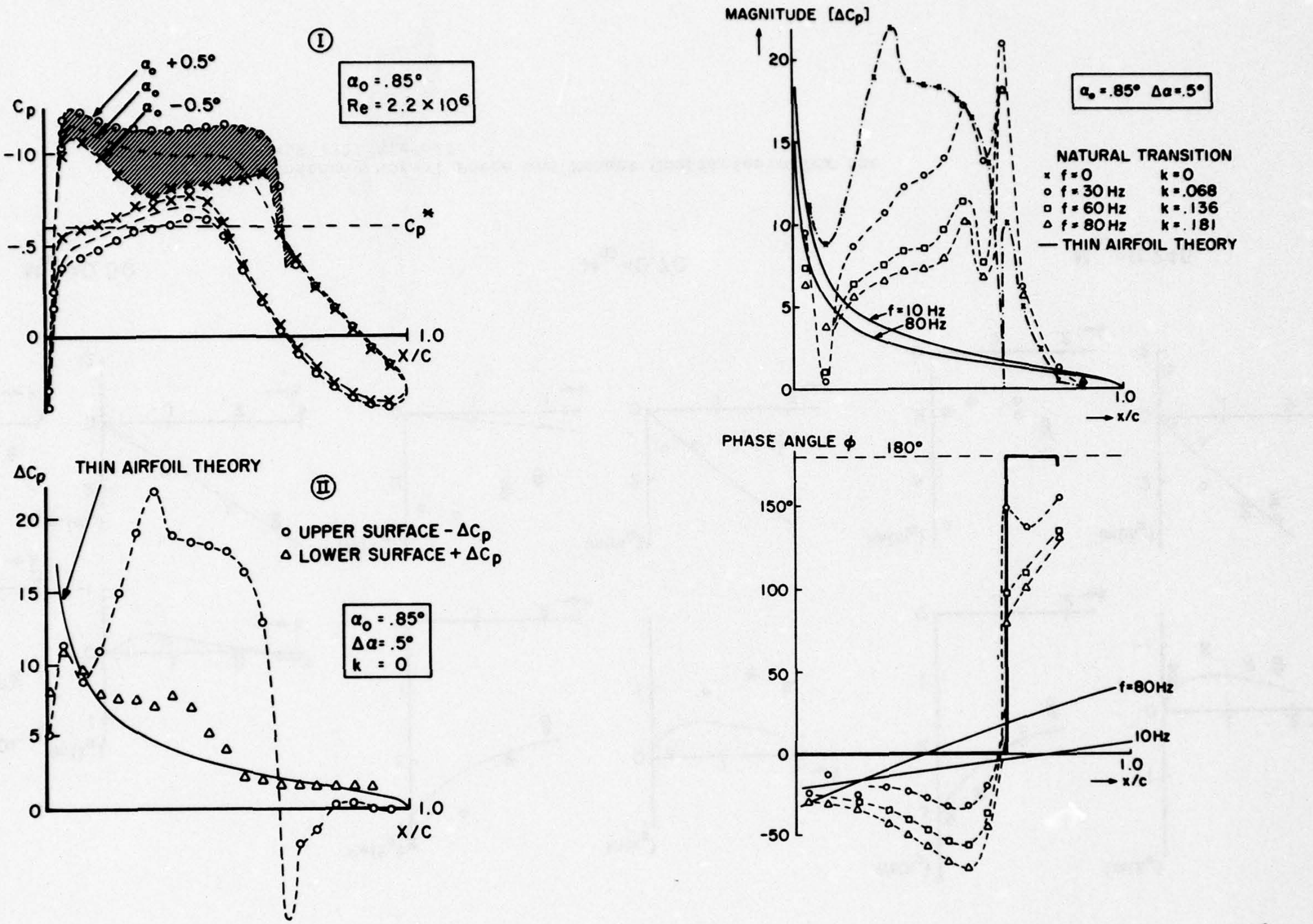


Fig. 8 Steady, Quasi-Steady, and Unsteady Pressures on an NLR 7301 Airfoil at M_{∞} =0.745.

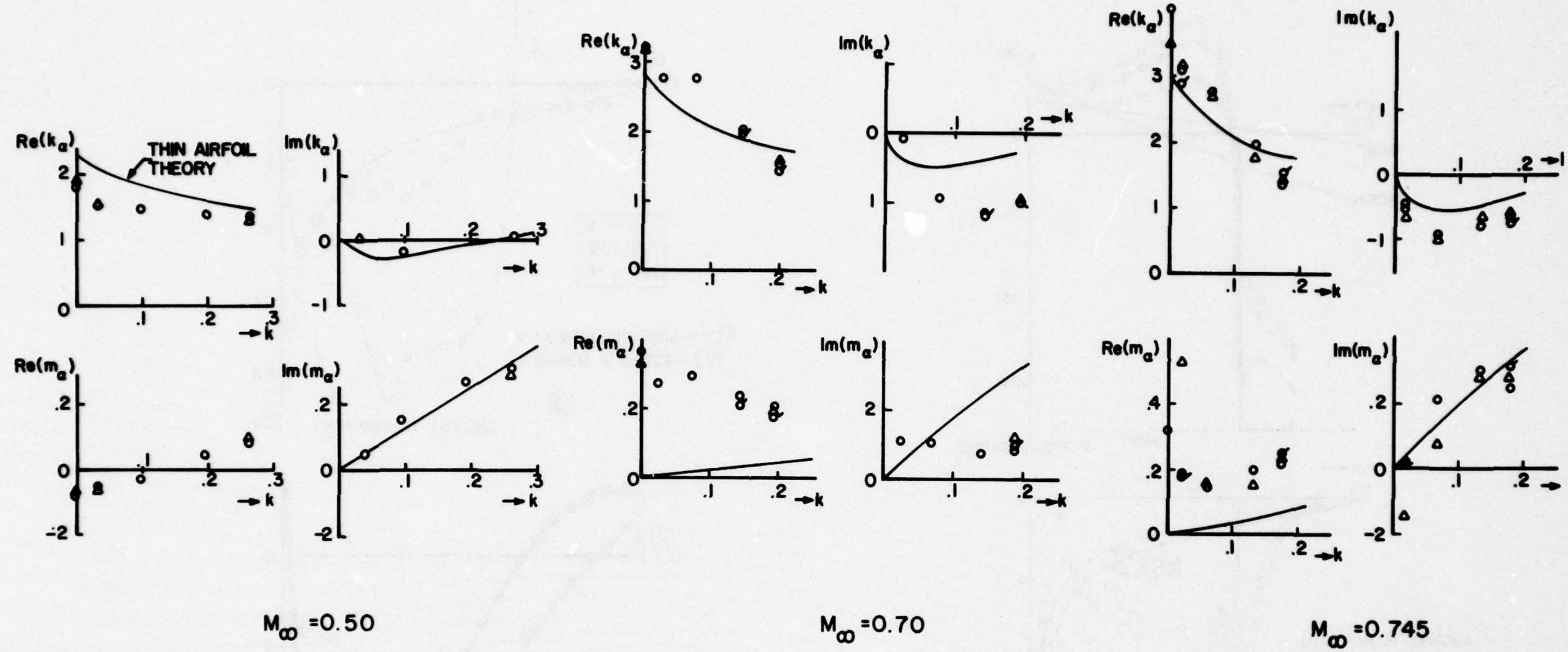


Fig. 9 Unsteady Normal Force and Moment Coefficients for the NLR 7301 Airfoil.

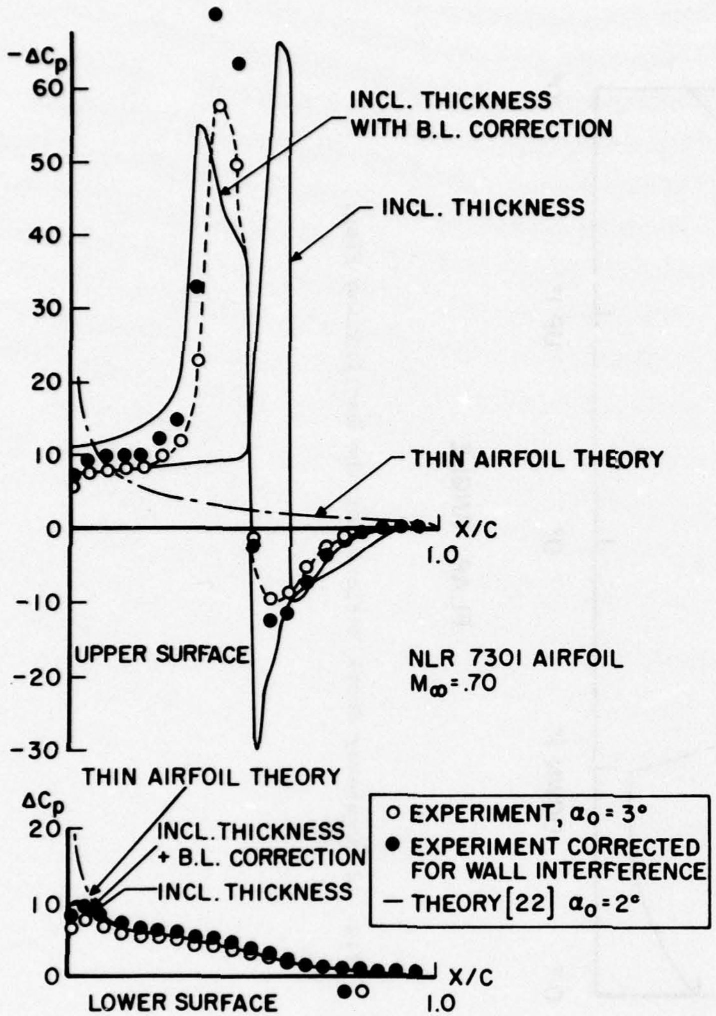


Fig. 10 Viscous and Thickness Effects on Quasi-Steady Transonic Pressures.

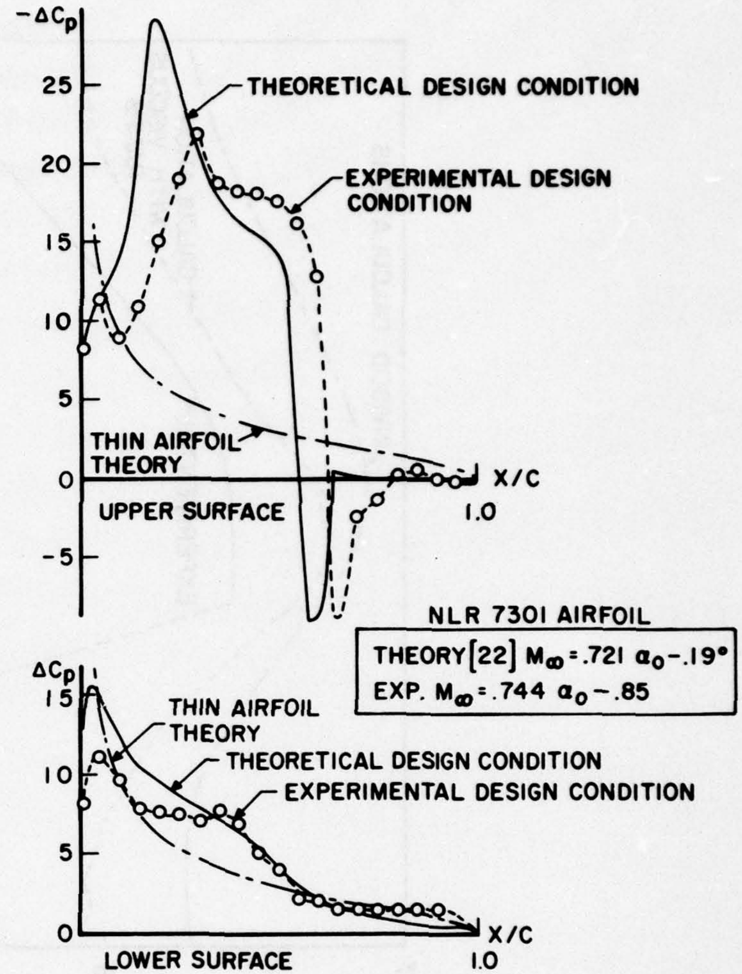


Fig. 11 Quasi-Steady Pressures for the "Shock-Free" Condition.

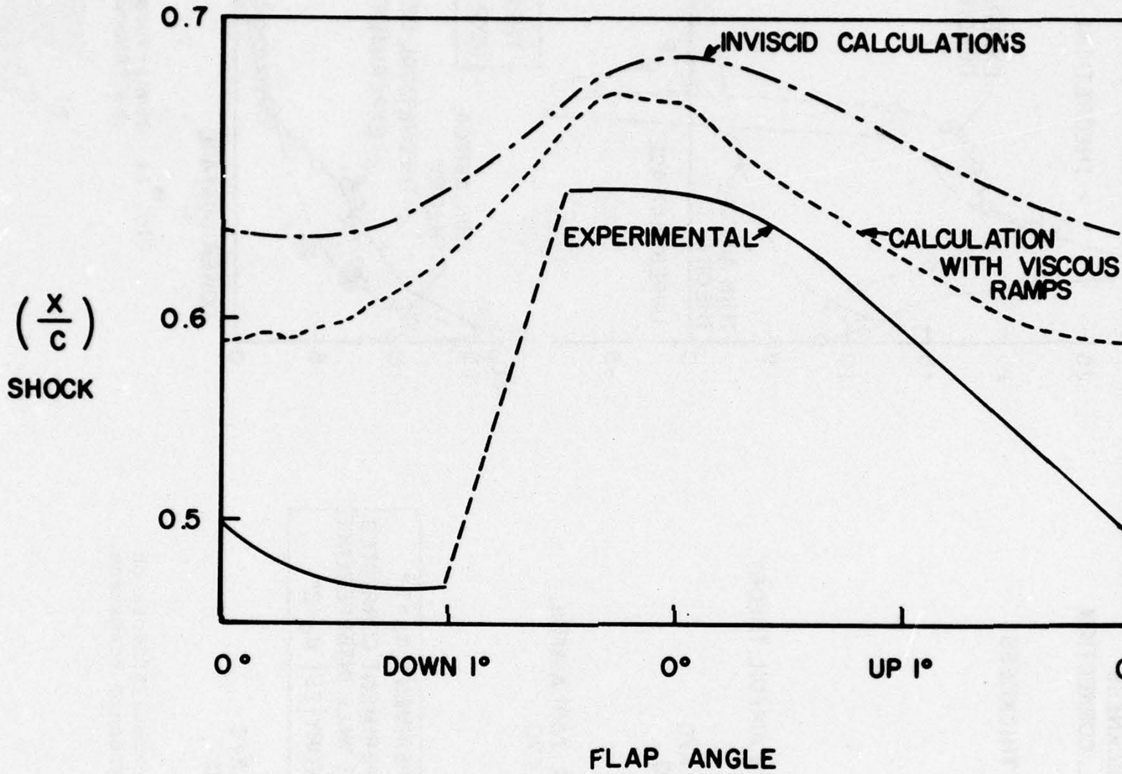


Fig. 12 Unsteady Shock Motions for the Oscillating Flap.

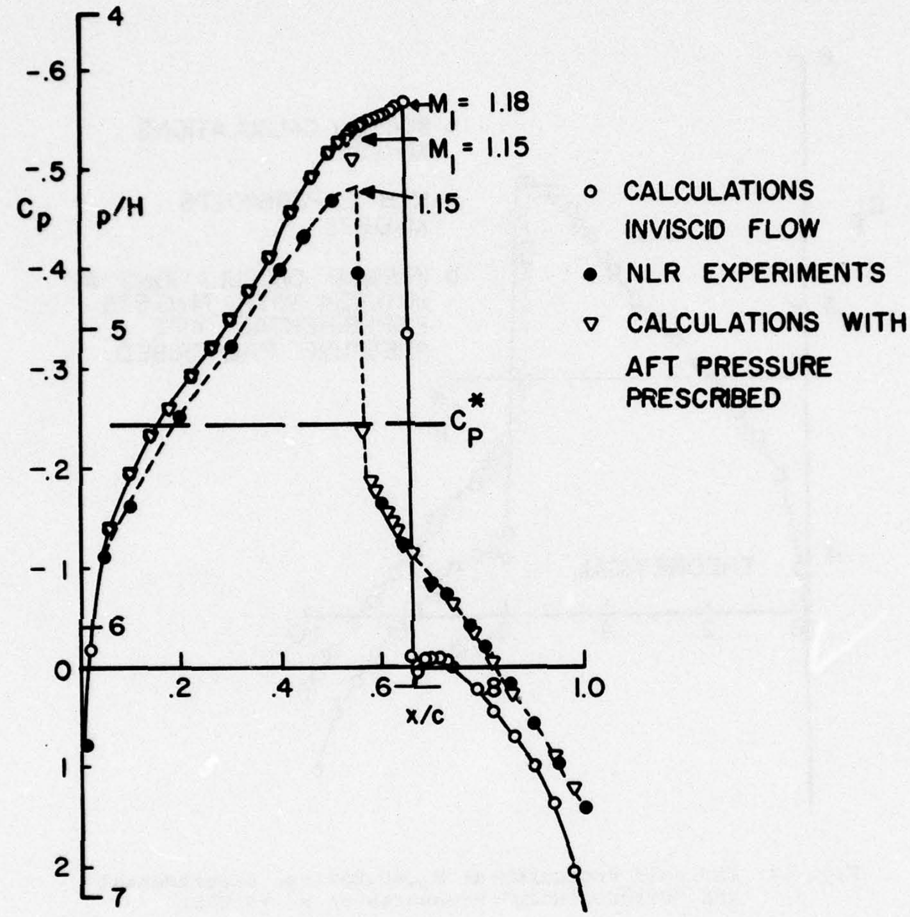


Fig. 13 Experimental, Inviscid and "Viscous-Ramp" Pressures for M_∞=0.875.

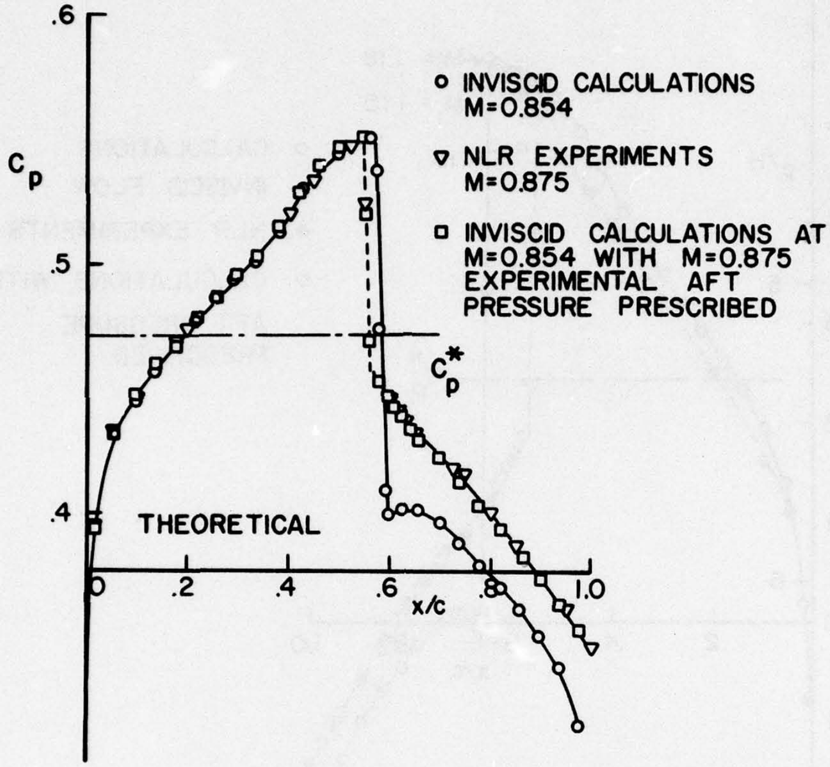


Fig. 14 Inviscid Pressures at M_∞=0.854 vs. Experimental and "Viscous-Ramp" Pressures at M =0.875.

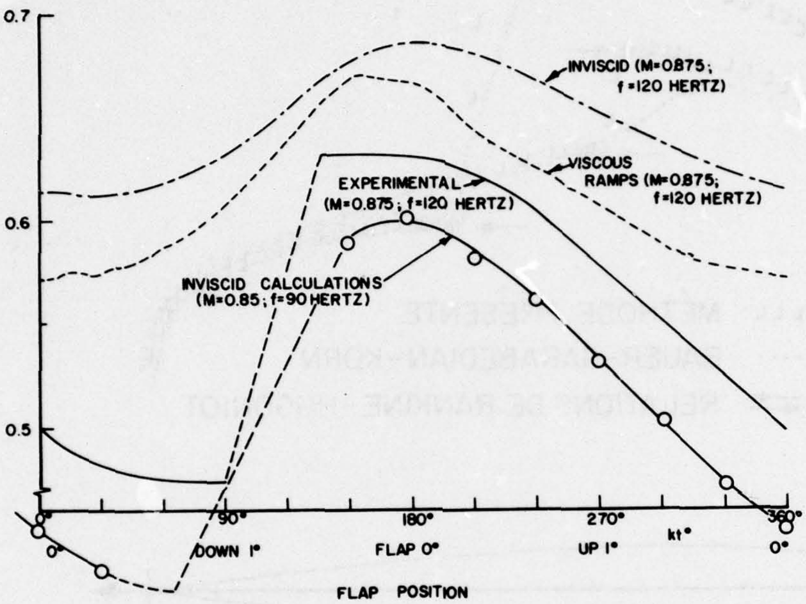


Fig. 15 Unsteady Shock Motions for the Oscillating Flap.

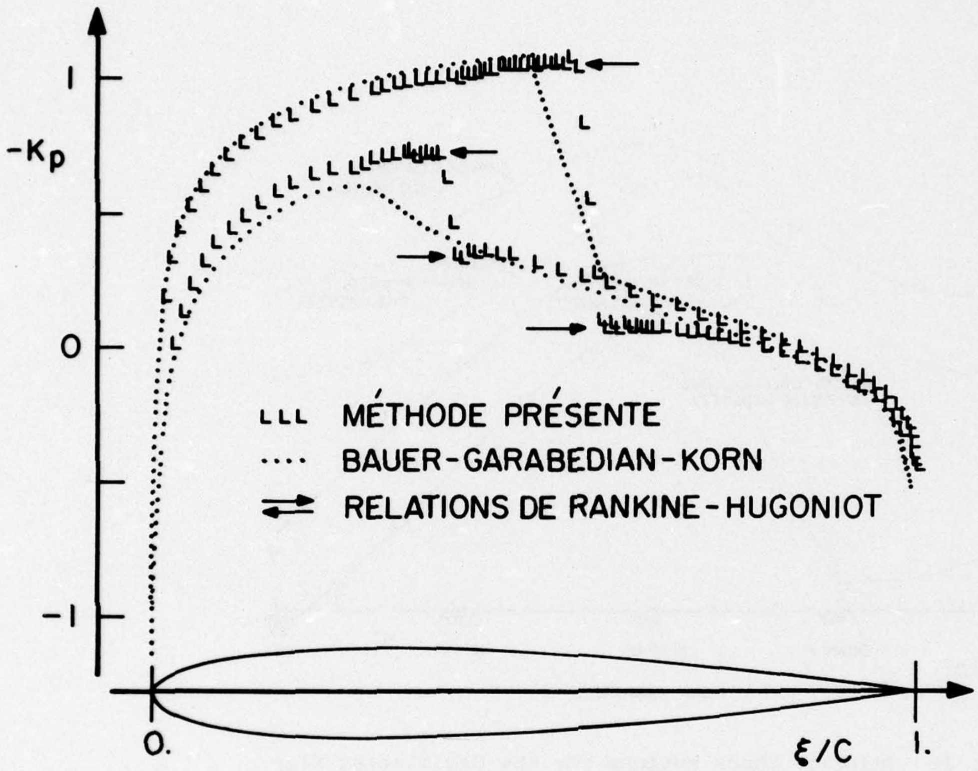


Fig. 16 Steady Pressures at $M_{\infty}=0.8$.

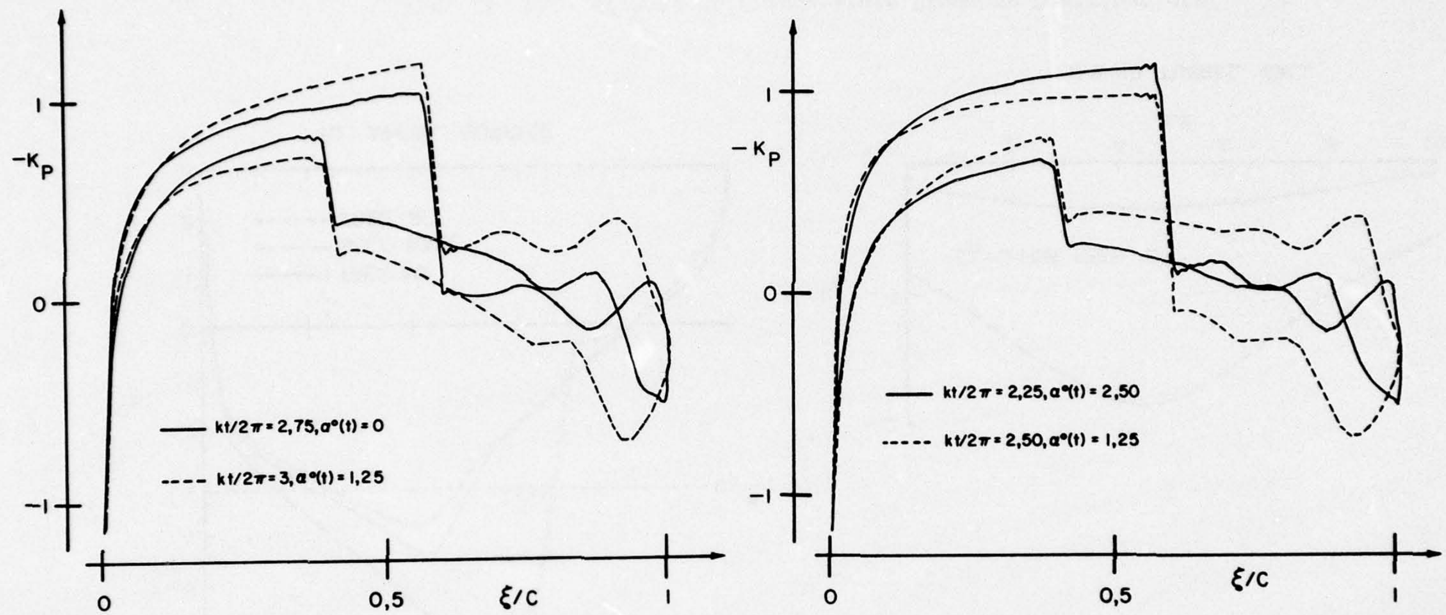


Fig. 17 Unsteady Pressures at M_∞=0.8, k=10.0.

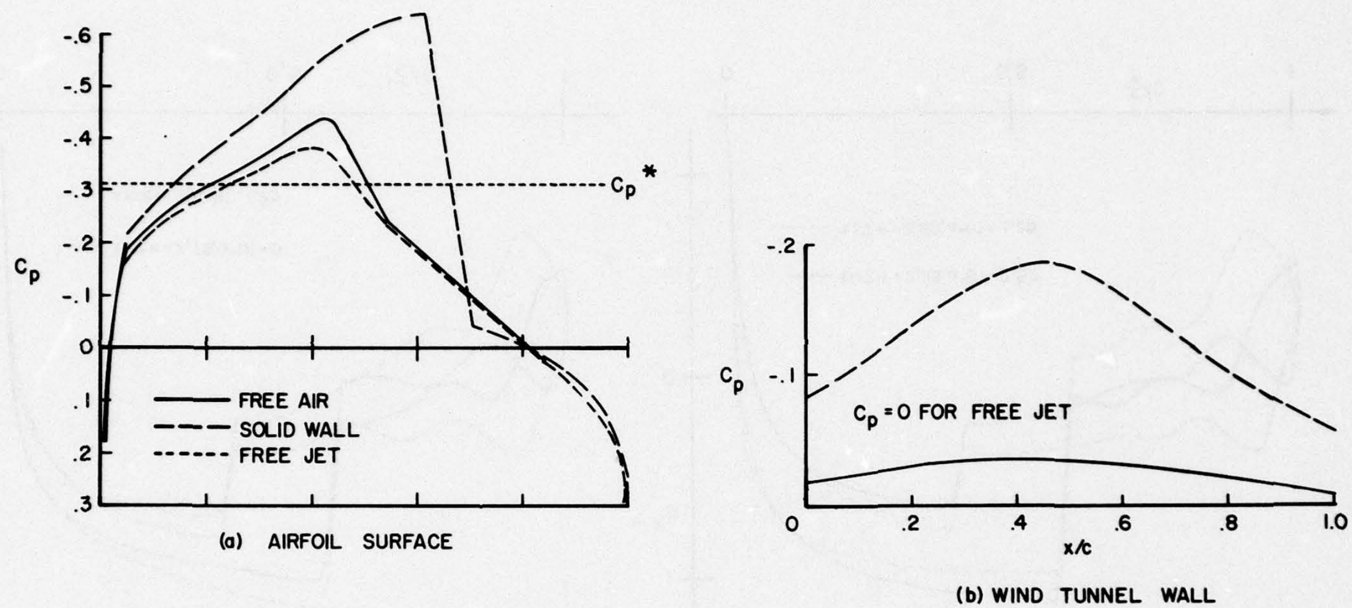


Fig. 18 Wall Effects on Steady-State Pressure Distributions.

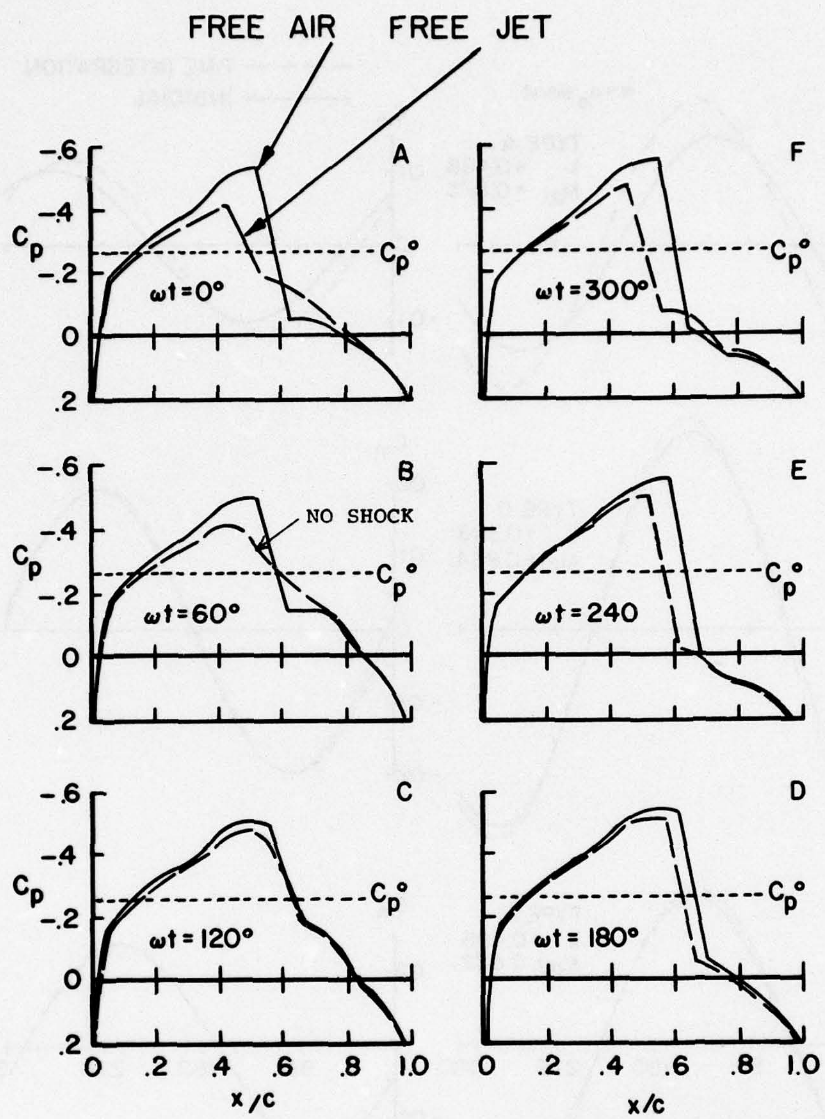


Fig. 19 Wind Tunnel Wall Effects on Unsteady Pressures, Oscillating Flap, $M_\infty = 0.865$, k=0.468.

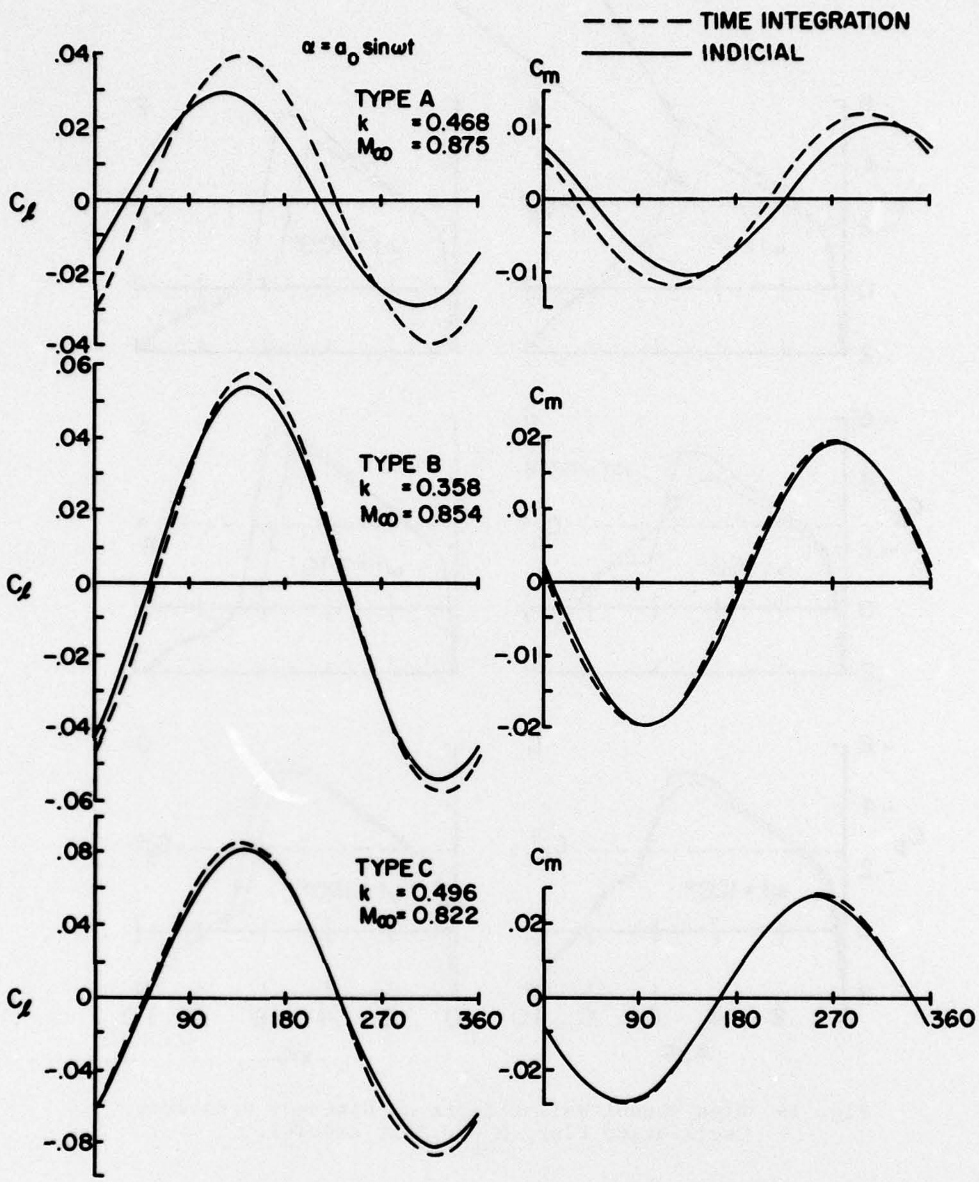


Fig. 20 Direct Integration and Indicial Calculation and Transient Lift and Moment.

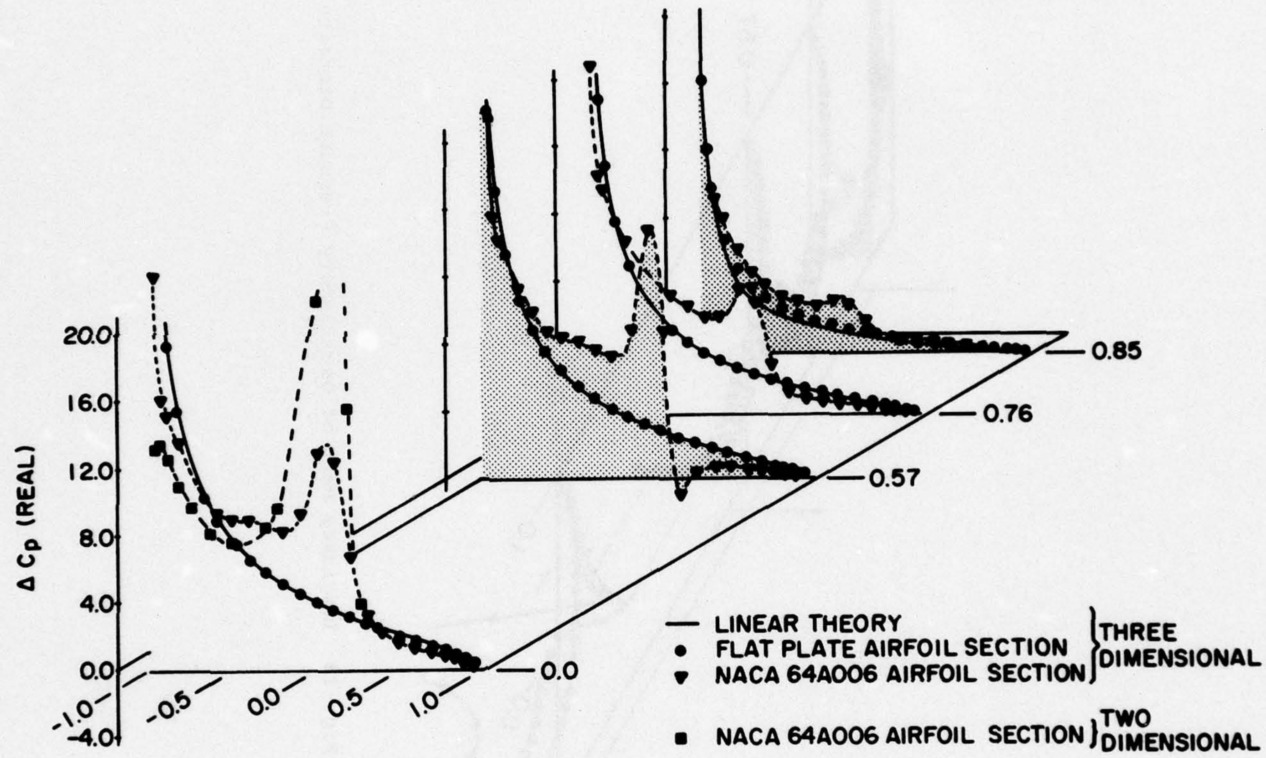


Fig. 21 Real Part of Oscillatory Pressure Distribution.

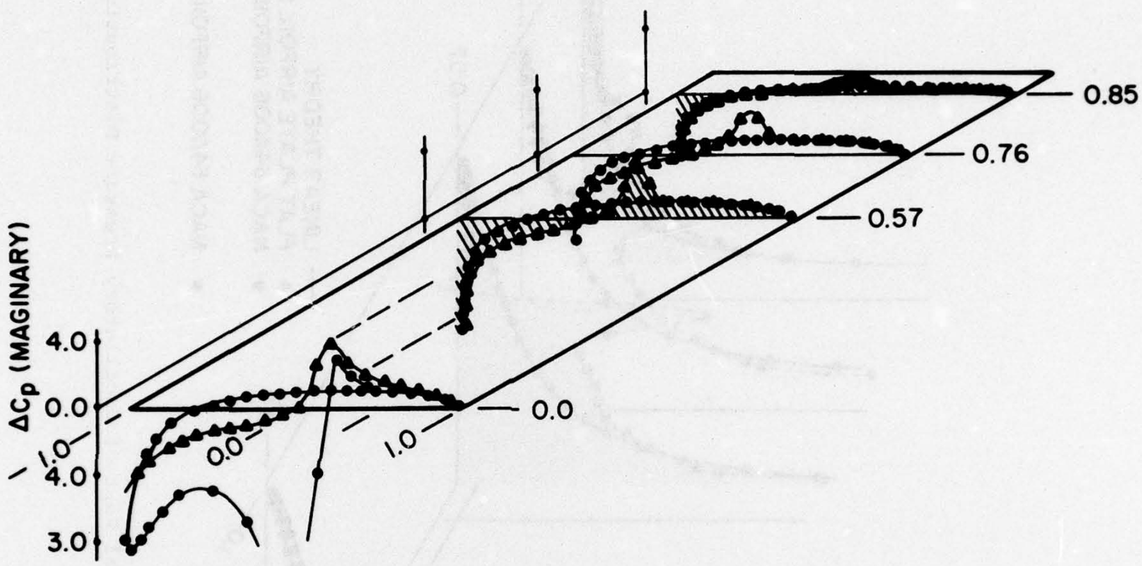


Fig. 22 Imaginary Part of Oscillatory Pressure Distribution.

Table 2. Comparison of Relaxation, Direct Inversion, and Exact Solutions of a One-Dimensional Problem.

x_j	ϕ_j RELAXATION	ϕ_j DIRECT INVERSION	ϕ_j EXACT
0.0	0.5000	0.5000	0.5000
0.1	-0.0025	-0.0036	-0.0084
0.2	-0.5195	-0.5065	-0.5151
0.3	-0.9428	-0.8969	-0.9094
0.4	-1.1776	-1.0881	-1.1055
0.5	-1.1667	-1.0377	-1.0606
0.6	-0.9067	-0.7568	-0.7845
0.7	-0.4518	-0.3078	-0.3374
0.8	0.0980	0.2094	0.1823
0.9	0.6204	0.6802	0.6640
1.0	1.0000	1.0000	1.0000

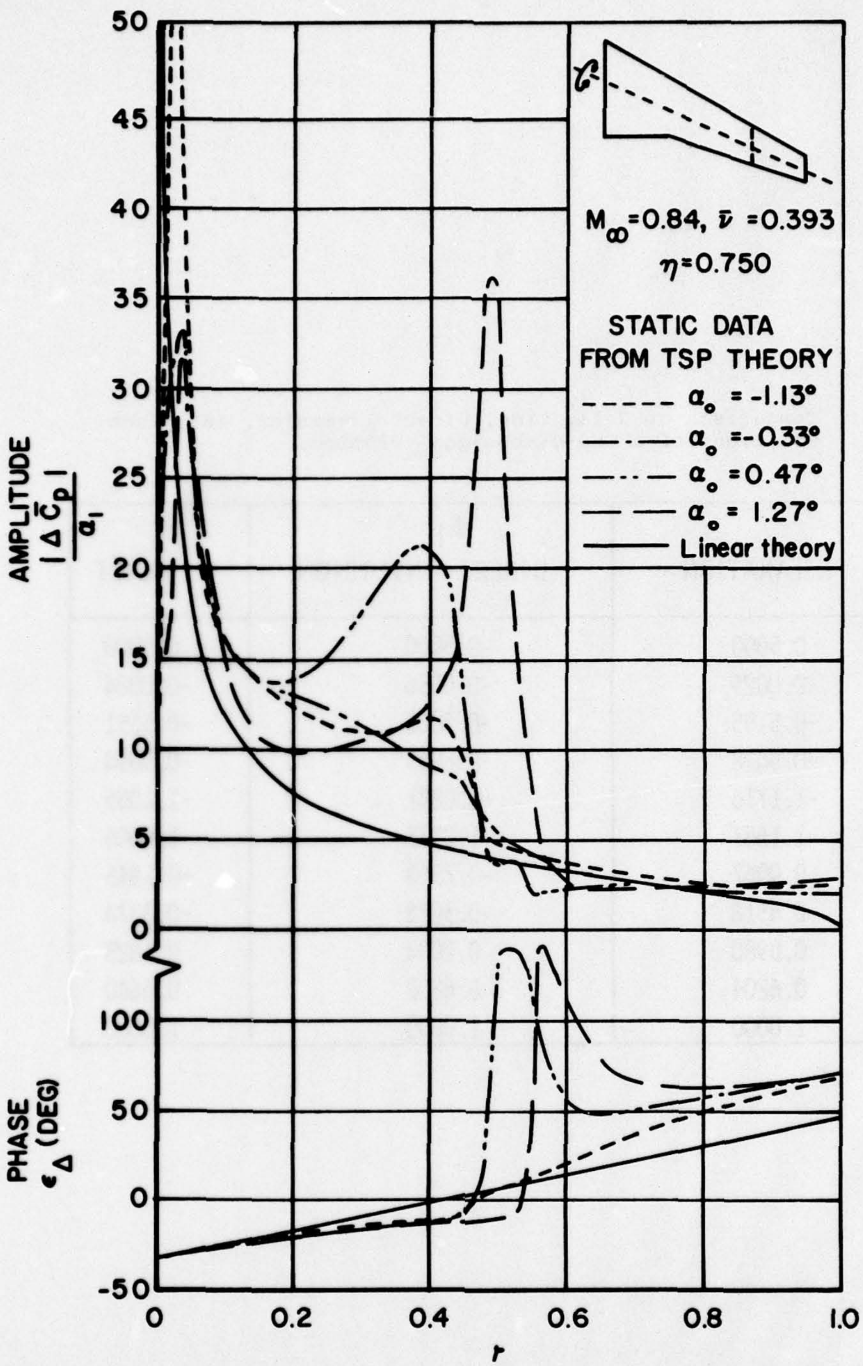


Fig. 23 The Effect of Initial Incidence on Calculated Oscillatory Pressure Distributions.

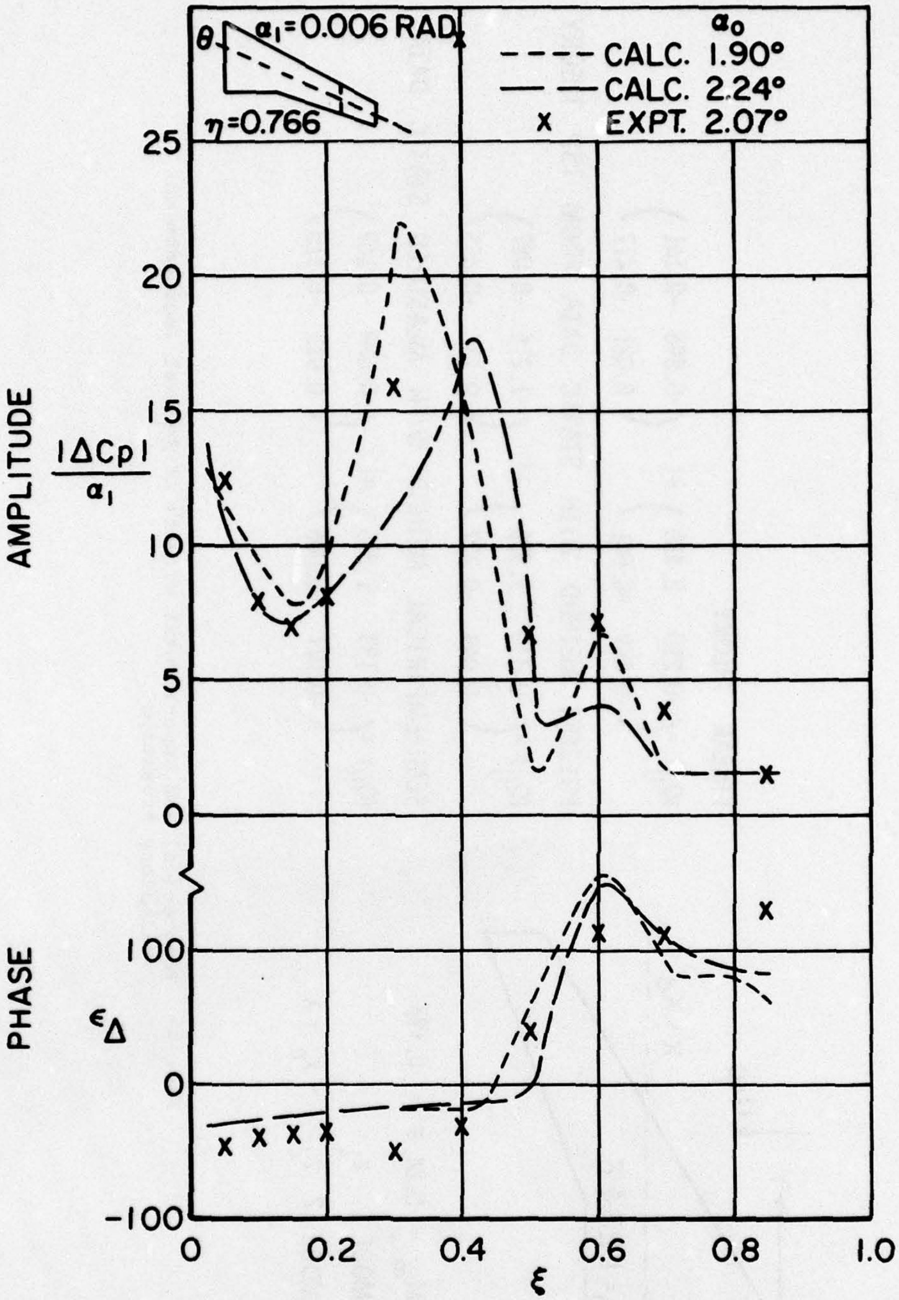
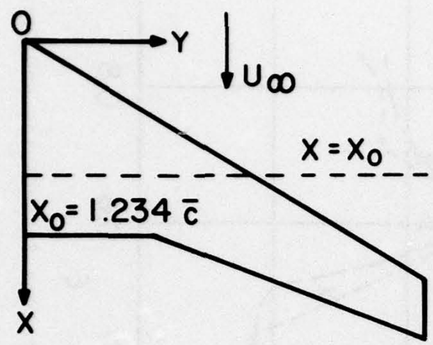


Fig. 24 Predicted Aerodynamic Forces for Heaving and Pitching Motion.



$M_\infty = 0.84, \bar{\nu} = 0.393$

MODE 1 $z_1 = \bar{c}$

MODE 2 $z_2 = x_0 - x$

LINEAR THEORY

$$(Q_{ij}) = \begin{pmatrix} -0.210 & 2.480 \\ -0.049 & -0.041 \end{pmatrix} + i \begin{pmatrix} -0.896 & -0.141 \\ 0.021 & -0.417 \end{pmatrix}$$

PRESENT METHOD WITH STATIC DATA FROM TSP THEORY

$$(Q_{ij}) = \begin{pmatrix} -0.338 & 3.526 \\ -0.043 & -0.327 \end{pmatrix} + i \begin{pmatrix} -1.276 & -0.087 \\ 0.137 & -0.455 \end{pmatrix}$$

SEMI-EMPIRICAL METHOD WITH MEASURED STATIC DATA

$$(Q_{ij}) = \begin{pmatrix} -0.133 & 1.895 \\ -0.077 & -0.006 \end{pmatrix} + i \begin{pmatrix} -0.689 & -0.169 \\ 0.015 & -0.310 \end{pmatrix}$$

Fig. 25 Theoretical and Experimental Effects of Initial Incidence on Oscillatory Pressures.

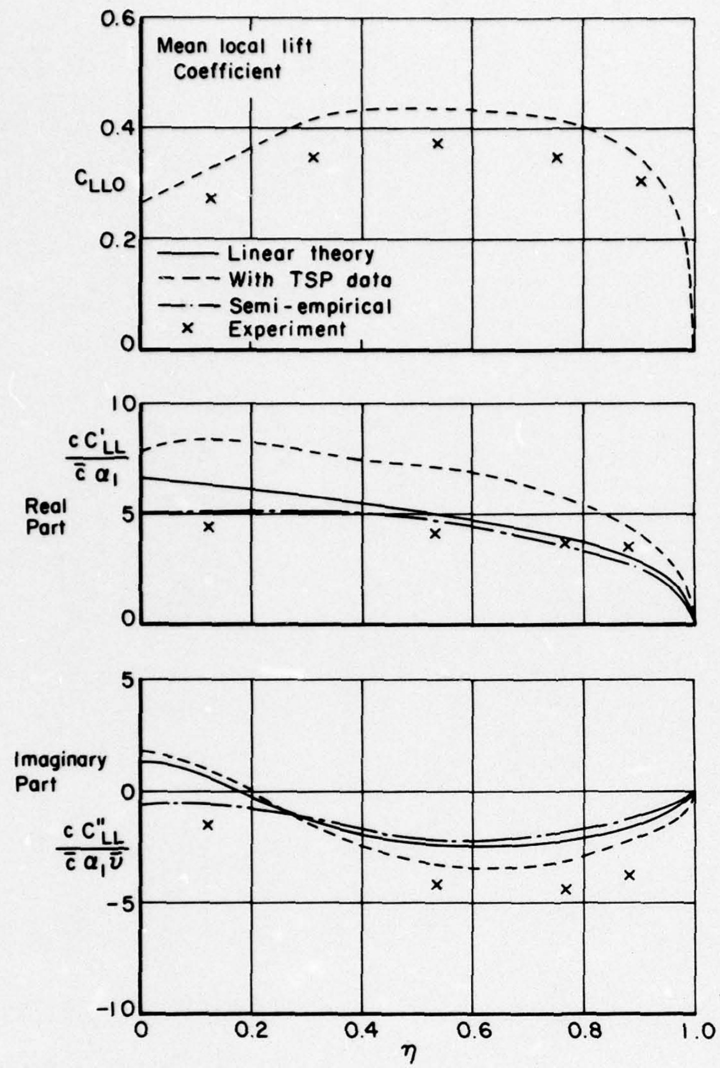


Fig. 26 Spanwise Distributions of Mean and Oscillatory Lift Section Lift Coefficients.

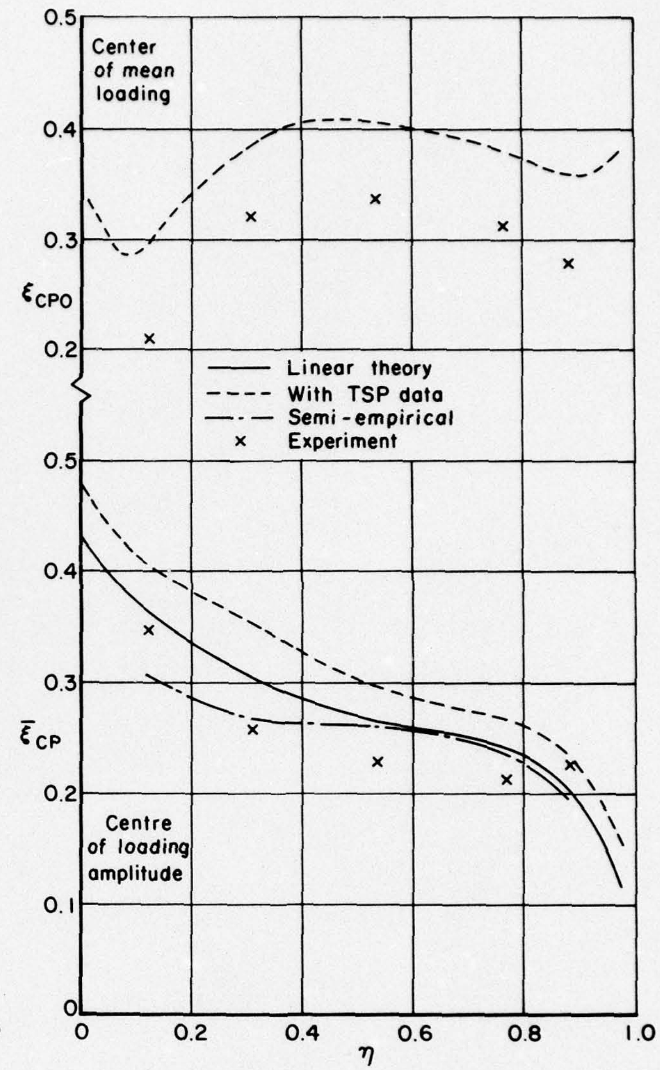


Fig. 27 Spanwise Distributions of Mean and Oscillatory Chordwise Centres of Pressure (M_∞ =0.84, ν =0.393)

APPENDIX

"COMMENTS ON THE STATE OF THE ART OF TRANSONIC UNSTEADY AERODYNAMICS"

H.C.Garner

During his lecture on "A Practical Framework for the Evaluation of Oscillatory Aerodynamic Loading on Wings in Supercritical Flow," Mr H.C.Garner presented a Venn diagram on the state of the art and additional comments which are not in his paper. Considerable interest was expressed in this information and, with his kind permission, they are reproduced in this appendix.

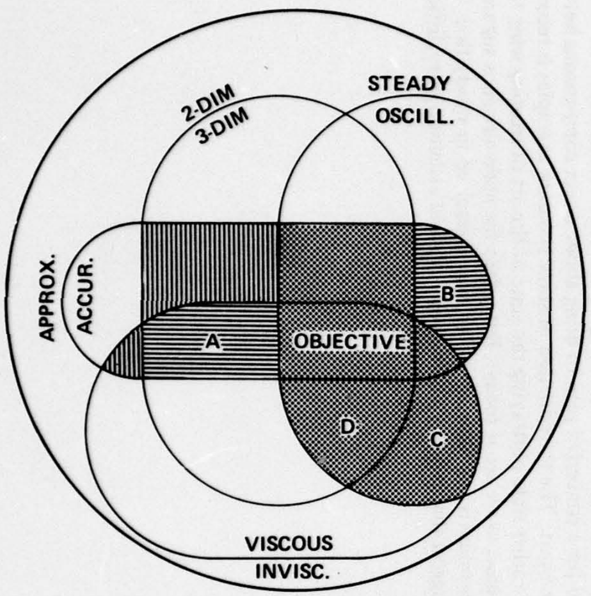
Please notice that "by *accurate solution*, he means one that would be expected to fill practical requirements, while the approximate solution is expected to fall short of this goal in mixed flow."

His additional comments are as follows:

"First, however, let me depict the state of the art in a Venn diagram (Plate 1). By supercritical flow I refer to a mean velocity field which is subsonic remote from the wing, but partly supersonic on its surface. In the shading scheme, denotes negligible progress; , a promising start; , solid development; and , successful conclusion. The outermost area corresponds to past successes in the approximate solution of two-dimensional, steady, inviscid flow, and the central objective of future research is the accurate solution of three-dimensional, oscillatory, viscous flow. While steady-state theory for supercritical flow is already well-developed, the problems of oscillatory viscous flow have scarcely been touched. The areas are those of greatest current effort; three-dimensional, steady, viscous flow in region A, and two-dimensional, oscillatory, inviscid flow in region B. Through the work of Dr. Ballhaus and M.Sides, region B should now be upgraded to , solid development. The region C expresses the need for fundamental work on two-dimensional, oscillatory, viscous flow, but this may be upgraded to (promising start), at least, on the basis of Dr. Yoshihara's and Mr Tijdeman's papers. My own contribution is to region D, and I hope to persuade you that this area of three-dimensional, oscillatory, viscous flow may also be changed to (promising start). Perhaps today's meeting enhances the Venn diagram (Plate 2), like that.

In conclusion I offer two general thoughts. The growing current effort and interest in unsteady three-dimensional transonic flow is creating a demand for a concerted plan to bring about direct comparisons between the results of the various theories that are being developed. The introduction of some standard examples is recommended sooner rather than later. (Plate 2) I return to an earlier slide portraying the state of the art of treating wing loading in supercritical flow when the main stream is subsonic, as we see it today. But, when the main stream is supersonic, (Plate 3) the problems of three-dimensional transonic viscous flow are by no means resolved in steady flow, but the oscillatory field is almost virgin territory. I suggest that military applications demand attention to realistic methods for low-supersonic flutter aerodynamics."

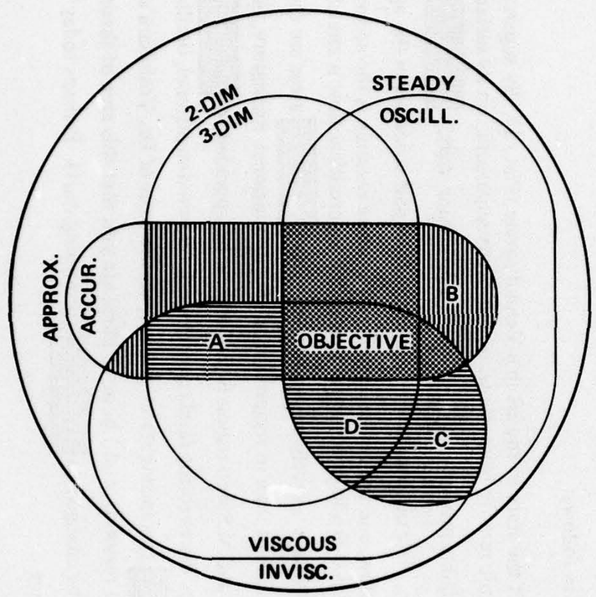
PLATE 1



Progress towards realistic solutions
for wing loading in supercritical flow

(BEFORE SPECIALISTS MEETING)

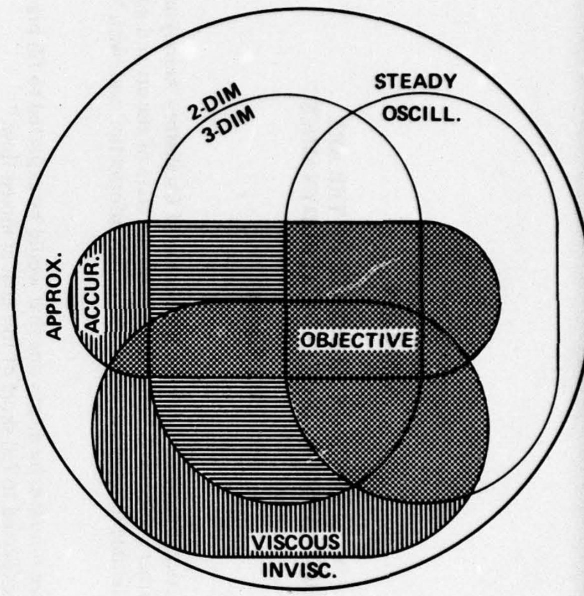
PLATE 2



Progress towards realistic solutions
for wing loading in supercritical flow

(AFTER SPECIALISTS MEETING)

PLATE 3



Progress towards realistic solutions
for wing loading in supersonic flow

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7. Presented at

8. Author(s)

W.J.Mykytow*, B.Laschka† and J.J.Olsen*

9. Date April 1978

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14. Abstract

The Specialists' Meeting, held on 19–20 April 1977, consisted of two Sessions on "Airframe Response to Separated Flow" and "Transonic Unsteady Aerodynamics for Aeroelastic Phenomena" respectively. The first Session reviewed the prediction and description of the separated flow environment and the essential effects of airframe response on individual aircraft components. The second Session dealt with flutter, aeroservoelastic instabilities involving coupling with active control systems and other static and dynamic aeroelastic problems, with specific reference to the transonic speed range.

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